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SHUTTLE VEHICLE AND MISSION

SIMULATION REQUIREMENTS REPORT

VOLUME I

10/20/72

SINGER
SIMULATION PRODUCTS

A DIVISION OF THE SINGER COMPANY • DEVELOPER AND MANUFACTURER OF THE LINK TRAINER SINCE 1929

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SIMULATION REQUIREMENTS REPORT

VOLUME I

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Principal Investigator
SMS Definition Study

This document is submitted in compliance with Line Item No. 2 of the Data Requirements List as Type I Data, Contract NAS9-12836

SINGER COMPANY
SIMULATION PRODUCTS DIVISION

SHUTTLE VEHICLE AND MISSION
SIMULATION REQUIREMENTS REPORT

VOLUME I

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PREFACE

This document is submitted in compliance with Line Item No. 2 of the Data Requirements List as Type I Data, Contract NAS9-12836. The document is divided into four volumes for ease of handling. The contents of each volume is defined as:

Volume I: Includes sections entitled Introduction, Mission Envelope and Flight Dynamics which correspond to Sections 1.0, 2.0 and 3.0 of the Table of Contents.

Volume II: Includes sections entitled Introduction and Shuttle Vehicle Systems which correspond to sections 1.0 and 4.0 to 4.18 of the Table of Contents.

Volume III: Includes sections entitled Introduction and Shuttle Vehicle Systems which correspond to sections 1.0 and 4.19 to 4.22 of the Table of Contents.

Volume IV: Includes sections entitled External Interfaces, Crew Prodedures, Crew Station, Visual Cues and Aural Cues which correspond to sections 5.0, 6.0, 7.0, 8.0 and 9.0 of the Table of Contents.

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4.17 Onboard Computer System

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1.0 Introduction

The objective of the Shuttle Vehicle and Mission Simulation Requirements report is to provide to NASA/MSC documentation of the requirements for faithful simulation of the Shuttle Vehicle, its systems, mission, operations and interfaces. To accomplish this objective the report was divided into eight topics which comprehensively cover the simulation requirements of the Shuttle mission and vehicle. The topics and their main objectives are summarized below.

Mission Envelope - This topic covers the space and atmospheric missions that are envisioned for the Shuttle program. The characteristics of each mission are described by an analysis of the mission phases, trajectory information, timelines and operations for nominal and abort conditions to the extent data was available.

Orbiter Flight Dynamics - This topic covers the flight regimes which the Shuttle vehicle will encounter in the accomplishment of its missions. The requirements were established in the following manner.

The vehicle configurations that must be simulated for horizontal and vertical test flights, operational space missions, atmospheric missions and abort modes were defined.

The dynamics requirements were established by defining the forces and moments that will act on the vehicle during the entire mission envelope which include, propulsion, gravity, aerodynamic effects, payload effects, docking effects, staging effects, ground reactions and the dumping of material overboard. The translational equations of motion requirements were established by defining the vehicles, satellites and payloads whose state vectors must be calculated and by defining the coordinate systems, relative equations of motion and accuracy of the calculations. A similar analysis was performed for the rotational equations of motion. Mass property and ephemeris requirements were also identified.

Shuttle Vehicle Systems - The Shuttle vehicle systems required for simulation were identified and described. The descriptive data generated in this effort was primarily based on the North American Shuttle proposal. The Shuttle vehicle and its system configuration is currently in a state of flux and therefore the descriptive data

contained in this report undoubtedly will become out of date as the Shuttle program progresses. However, for the purposes of this study, the data is more than adequate to define simulator requirements and a baseline design when it is tempered with the past experience of Apollo and Gemini programs. A cross correlation between the NR definition of systems and LRU's and this report is shown in Table 1-1 for reference purposes.

External Interfaces - The external interfaces of the Shuttle vehicle were identified and a preliminary type interface description established. Due to the fact that for every external interface there also exists an equivalent on-board system, the descriptive data on the workings of the interfaces is contained in the Shuttle Vehicle Systems section of the report and cross references are provided in this section.

Crew Procedures - The actual crew procedures for the Shuttle system will not be available for many years. As a result the study concentrated on identifying tasks by mission phase and crew member and identifying the probable interfaces between work stations. The data used for the

analysis was a RTOP study by MacDonald-Douglas, conversations with George Franklin of NASA/MSC, past experience, and the requirements of the Shuttle vehicle & mission.

<u>Crew Station</u> - The latest available data at the time of the writing of this report was used to identify the configuration of the Crew Station. The shape of the interior cabin, the location of the work stations and the allocation of the C&D panels by work station were established. Detailed data on the interior composition of the cabin is not currently available. However, simulation requirements were identified based on past experience and accepted levels of fidelity for mission simulators.

<u>Visual Cues</u> - The visual scene content was established for each of the mission phases. Attributes of the scene elements, to the extent feasible, were established and will be further defined in the SMSR report. The vehicle window configuration is not defined at this time but the best data available was utilized. The accelerations, velocites and displacements were established to the extent possible. Some

dynamics data was not available such as in the Abort phases of the mission. The missing information will be incorporated if it becomes available when the time frame and ground rules of the study or assumptions will be made.

<u>Aural Cues</u> - The aural cues requirements associated with the mission and vehicle systems were identified and described. Detailed data on the characteristics of each sound was not available and probably will not be until the vehicle test program is in progress. This factor can be circumvented by specifying flexibility into the simulator aural cue equipment.

This report will be updated at the end of the study based on data received as of January 1, 1972.

Reference to study data sources are included in the margins and the text in order to facilitate update of this report. The numerical references are correlated with the data listing defined by Table 1-2.

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TABLE 1-1
SPACE SHUTTLE ON-BOARD EQUIPMENT **CROSS REFERENCES**

SYSTEM: AVIONICS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	/Remarks/Assumptions
Star Sensor	3	4.9	ITT Model used on Aero Bee but does not meet proposed specs. Specs. and data required.
Rate Sensor Package	3	4.9	Honeywell GG 1027 Model used on F-14 AFCS. Data and Specs. required
Angle of Attack Transducer	3	4.9	Honeywell HG 280 used on DC 10.
IMU	3	4.9	Singer model KT70 used on A7D/E.
IMU Power Supply	3	4.9	Singer model KT70 used on A7D/E.
TVC Monitor	2 (?)	4.9.	No Data Exact function not known.
Air Data Package	3 (?)	4.9. 4.9.	Honeywell Model HG280 used on DC10.
MPS TVC Drivers	3	4.3. 4.9.	No Data Available
Manual TVC/RCS Control	1	4.9.	Honeywell Model BG 286 used on Apollo SCS.
Aero Control Electronics Unit	?	4.9.	Honeywell AFCS used on F-14.
Horizon Sensor Assembly	3	4.9.	Barnes Model 15-163
OMS/TVC Driver Unit	3(?)	4.9.	No Data Available

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TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: AVIONICS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks /Assumptions
APS Driver/Monitor	3	4.9.	Honeywell Model BG.287 used on Apollo SCS.
Accelerometer Package	3	4.9.	Honeywell Model G.G.1026 used on F-14 AFCS
Aero Back-up Electronics	1	4.9.	No Data available
Subsystem Sequence Controller	2(?)	4.9	To be used for unmanned flights. No data available
Gyro Accelerometer Package	1	4.9.	No Data Available
Backup Optical Unit	1	4.9.	Apollo COAS
Throttle/Speed Brake Electronics	?		No Data
GN &C Computer	3(?)	4.1.8.3	IBM Model AP101 or Singer/Kearfott SKC2000.
Program I/O Processor	(?)		IBM SP1
FDAI/EDA	(?)		Honeywell JG 264/BG 285 used on Apollo SCS.
FCS Control Panel	(?)		Honeywell F-14

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: ELECTRICAL POWER

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
BATTERY	2	4.1 ELECTRICAL POWER	NICKEL-CADIUM - 10 AMPHOUR - 28 VOLT
GENERATOR CONTROL UNIT	3	4.1 ELECTRICAL POWER	APU DRIVEN GENERATOR
TRANSFORMER RECTIFIER UNIT	3	4.1 ELECTRICAL POWER	150 AMP
REMOTE CONTROL CIRCUIT BREAKER	?	4.1 ELECTRICAL POWER	MAGNETIC LATCH - HERMETIC SEALED UNITS
REMOTE POWER CONTROLLER	4	4.1 ELECTRICAL POWER	MAGNETIC LATCH - HERMETIC SEALED UNITS
BATTERY CHARGES	1	4.1 ELECTRICAL POWER	CONSTANT CURRENT CHARGER - DUAL REDUNDANT OUTPUT
INVERTERS	4	4.1 ELECTRICAL POWER	30, 1250 VA, 115/200V, 400 HZ
SEQUENCERS	2	4.1 ELECTRICAL POWER	NO DATA AVAILABLE
CONTROL TRANSFORMER RECTIFIER	?	4.1 ELECTRICAL POWER	NO DATA AVAILABLE
FUEL CELL	3	4.1 ELECTRICAL POWER	7/10 KW RESTARTABLE - CRYOGENIC 02 and H2 30 VOLT OUTPUT
ALTERNATOR - GENERATOR	3	4.1 ELECTRICAL POWER	20/30 KVA APU DRIVEN SPRAY OIL COOLED WITH INTEGRATED GEARBOX
FUEL CELL HEAT EXCHANGER	3	4.1 ELECTRICAL POWER	

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: MECHANICAL POWER

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
AUXILLARY POWER UNITS	4	4.2.1 AUXILIARY POWER UNITS	200 HORSEPOWER - USES HYDRAZINE
HYDRAZINE TANK	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVAILABLE
HELIUM TANK	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVAILABLE
HEAT EXCHANGER	4	4.2.1 AUXILIARY POWER UNITS	NO DATA AVILABLE
HYDRALIC PUMPS	8	4.2.2 HYDRALIC POWER SYSTEM	3000 PSI
ALTERNATOR	3	4.2.1 AUXILIARY POWER UNITS	400 HZ, 30, 20/30 KVA

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TABLE 1-1
SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: OPERATIONAL INSTRUMENTATION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
PILOT VOICE RECORDER	1	4.11.1 RECORDERS	
SWITCH SCAN MULTIPLEXER	12	FIGURE 4.11-1	
CAUTION AND WARNING	2	FIGURE 4.11-1, 4.11.4	AUTONETICS - APOLLO TYPE (NEW ITEM)
CRASH RECORDER	1	4.11.1 RECORDERS	SUNSTRAND, ECHO SCIENCE, OR DAVOLL FERRY USE ONLY
SIGNAL CONDITIONING UNIT-DFI	17	4.11.2 SENSORS AND SIGNAL CONDITIONING	SAT/APOLLO AUTONETICS SCE
TIMING UNIT (MTU)	2	FIGURE 4.11-1	APOLLO CTE, GENERAL TIMC
LOOP RECORDER	1	FIGURE 4.11-1 4.11.1 RECORDERS	SUNDSTRAND, ECHO SCIENCE, OR DAVOLL (5 MINUTE PLAYBACK)
PCM RECORDER - PAYLOAD	1	FIGURE 4.11-1 4.11.1 RECORDERS	SUNDSTRAND, ECHO SCIENCE OR DAVOLL (MAINT. AND PAYLOAD)
OPER. TRANSDUCERS	2359 DFI 2803 DFI	FIGURE 4.11-1 4.11.2 SENSORS AND SIGNAL CONDITIONING	VARIOUS MAKES
PCM REMOTE UNIT DFI	1	FIGURE 4.11-1	SCI, TELEDYNE
PCM MASTER UNIT - DFI	2	FIGURE 4.11-1	DFI ONLY SCI, TELEDYNE
GROUND CHECKOUT DECODER	?	?	MAY NOT EXIST

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TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: OPERATIONAL INSTRUMENTATION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
GN&C COMPUTER 64K	2?	4.18.3	IBM MODEL AP101 OR SINGER/KEARFOTT SKC 2000
INPUT-OUTPUT BUFFER	?	4.18.2.9.3/ 4.18.2.9.4	SP-1 COMPUTER STRUCTURES SKYLAB POWER SUPPLY, AP1/SP1
MDE UNIT	?	4.19-4.19-7	IBM SP1
MAGNETIC TAPE READER	?	4.19.2	NO DATA AVAILABLE
TAPE CONTROL ELECTRONICS	?	4.19.2	NO DATA AVAILABLE
CRT DISPLAY UNIT	8?	4.19.2.1	IBM-F14 TYPE HEAD WITH ADDITION OF A READ/WRITE REFRESH BUFFER, A SYMBOL
			GENERATOR, ANALOG AND DIGITAL CONTROL LOGIC, D/A'S AND POWER SUPPLIES
DFI TIMING UNIT	1		NO DATA AVAILABLE
WIDEBAND RECORDER	1		NO DATA AVAILABLE
FREQUENCY MULTIPLEXER	3		NO DATA AVAILABLE
PCM RECORDER DFI	1		NO DATA AVAILABLE
PCM RECORDER MAINTENANCE	1		NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: OPERATIONAL INSTRUMENTATION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
PCM MASTER UNIT - OFI	2		NO DATA AVAILABLE
PAYLOAD DATA INTERLEAVER	2		NO DATA AVAILABLE
PCM REMOTE ACQUISITION UNIT	24		NO DATA AVAILABLE
DEDICATED SIGNAL CONDITIONER	6		NO DATA AVAILABLE
COMMAND ACQUISITION UNIT	1		NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: D&C

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
VERTICAL SPEED	2	Note: The SV&MSR did not address detailed D&C instruments due to the lack of firm data	BENDIX E-C, AAK-23/A24G-17A
BARO ALTITUDE	2	NO DATA AVAILABLE	AEROSONICS, AAU-16/A
IAS/MACH	2	NO DATA AVAILABLE	BENDIX E-C, ASK-14/A24G-18
FDAI (3 AXIS)	2	NO DATA AVAILABLE	MODIFIED APOLLO CM FDAI
HSI	2	NO DATA AVAILABLE	BENDIX E-C, ACA AQU-4A
TAS/SAT	1	NO DATA AVAILABLE	NO DATA AVAILABLE
ACCELEROMETER	2	NO DATA AVAILABLE	NO DATA AVAILABLE
LG POSITION	1	NO DATA AVAILABLE	3 DISPLAYS - LEFT, RIGHT, NOSE
RCS PRESSURE	3	NO DATA AVAILABLE	DOUBLE POINTER
OMS PC	1	NO DATA AVAILABLE	NO DATA AVAILABLE
OMS FUEL	1	NO DATA AVAILABLE	NO DATA AVAILABLE
OMS OX	1	NO DATA AVAILABLE	NO DATA AVAILABLE

TABLE 1-1
SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: D&C

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
EVENT TIMER	2	NO DATA AVAILABLE	NO DATA AVAILABLE
HYD. PRESSURE	1	NO DATA AVAILABLE	NO DATA AVAILABLE
MPS PC	1	NO DATA AVAILABLE	NO DATA AVAILABLE
MPS LH2/LO2	1	NO DATA AVAILABLE	NO DATA AVAILABLE
EXT. TANK QUANTITY	1	NO DATA AVAILABLE	NO DATA AVAILABLE
ISS DISAGREE	2	NO DATA AVAILABLE	NO DATA AVAILABLE
CMD DISAGREE	2	NO DATA AVAILABLE	NO DATA AVAILABLE
DRIVER FAIL	2	NO DATA AVAILABLE	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
SGLS INTERROGATOR	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
VHF TRANSCEIVER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
ATC TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SGLS TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SGLS DECODER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
USB TRANSPONDER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SIGNAL PROCESSOR	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
AUDIO CONTROL CENTER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
TACAN TRANSPONDER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
COMMAND DECODER	2	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
RADAR ALTIMETER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
WIDEBAND TRANSMITTER S-BAND	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
S-BAND ANTENNA	4	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP)
C-BAND ANTENNA	6	4.10 COMMUNICATIONS AND TRACKING	HORN (LP) FOR RADAR ALTIMETER
L-BAND ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	ANNULAR SLOT (VP) FOR TACAN AND ATC
UHF/VHF ANTENNA	3	4.10 COMMUNICATIONS AND TRACKING	HP DUAL CAVITY FOR ILS
VHF ANTENNA	2	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP)
VHF ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	TOP CAP (VP)
VHF ANTENNA	1	4.10 COMMUNICATIONS AND TRACKING	SPIRAL (VP)
L-BAND ANTENNA	2	4.10 COMMUNICATIONS AND TRACKING	HELIX IN CAVITY (RHCP) FOR TACAN
L-BAND ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
VHF ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
S-BAND ANTENNA SELECTOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
CCTV CAMERA (B&W)	4	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

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TABLE 1-1
SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: COMMUNICATION AND TRACKING

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
TV CAMERA - COLOR	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
DFI TRANSMITTER	1	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
ILS RECEIVER	3	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE
SECURE TERMINAL	4	4.10 COMMUNICATIONS AND TRACKING	NO DATA AVAILABLE

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TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: AIRBREATHING PROPULSION

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
TURBOFAN ENGINE	4	4.6 AIRBREATHING PROPULSION SYSTEM	P&W F401-PW-400 MODIFIED
FUEL TANK	1,2,or3	4.6 AIRBREATHING PROPULSION SYSTEM	22,565 LBS FOR EACH TANK

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: SOLID ROCKET MOTORS

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
MOTOR	2	4.7.1 MAIN SRM	SOLID PROPELLANT 156"
MOTOR	12	4.7.1 MAIN SRM	SOLID PROPELLANT 27K LBS @ 2 SECONDS
MOTOR	2	4.7.2 ABORT SOLID ROCKET MOTOR	SOLID PROPELLANT 385,000 LBS AVERAGE @ 21 SECONDS
MOTOR	2	4.7.3 DEORBIT SRM FOR EXTERNAL TANK	SOLID PROPELLANT 18,500 LBS @ 37 SECONDS

TABLE 1-1

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: EXTERNAL TANK

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
OXYGEN TANK - LIQUID	1	4.8.1 STRUCTURE	
HYDROGEN TANK - LIQUID	1	4.8.1 STRUCTURE	
BATTERY	2	4.8.4 AVIONICS	
ORDINANCE TIMING SYSTEM (DEORBIT AVIONICS)	1	4.8.4 AVIONICS	
RANGE SAFETY AVIONICS	1	4.8.4 AVIONICS	

TABLE 1-

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: MAIN PROPULSION SYSTEM

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
MPS ENGINE INTERFACE UNIT	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
MPS CONTROLLER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
FUEL PREBURNER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
OXIDIZER PREBURNER	3	4.3 MAIN PROPULSION SYSTEM	NO DATA AVAILABLE
MAIN ENGINE	3	4.3 MAIN PROPULSION SYSTEM	470K VAC THRUST
ENGINE ACTUATOR	6	4.3.5 ENGINE ACTUATOR	NO DATA AVAILABLE

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TABLE 1-

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: REACTION CONTROL SYSTEM

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
REACTION CONTROL MOTOR	40	4.4 REACTION CONTROL SYSTEM	1000 LBS THRUST VAC
HELIUM PRESSURIZATION SPHERES	4	4.4.4 PRESSURIZATION SUBSYSTEM	4000 PSI TANKS
MONOPROPELLANT HYDRAZINE FUEL TANKS	8	4.4 REACTION CONTROL SYSTEM	25 INCH DIAMETER X 42 INCH LONG CYLINDERS

TABLE 1-

SPACE SHUTTLE ON-BOARD EQUIPMENT CROSS REFERENCES

SYSTEM: ORBITAL MANEUVERING SYSTEM (OMS)

EQUIPMENT	NUMBER OF UNITS	SV & MSR Paragraph Number and Title	Remarks
ENGINE	2	4.5 ORBITAL MANEUVERING SYSTEM	5000 LBS VAC
TANK OXIDIZER	2	4.5 ORBITAL MANEUVERING SYSTEM	12,200 LBS MAX. - NITROGEN TETROXIDE
TANK FUEL	2	4.5 ORBITAL MANEUVERING SYSTEM	12,200 LBS MAX. - MONOMETHYLHYDRAZINE
HELIUM TANK	2	4.5 ORBITAL MANEUVERING SYSTEM	3200 PSIA

SORTED BY INDEX NUMBER

DATA REFERENCES
SHUTTLE MISSION SIMULATOR STUDY

OCTOBER 20, 1972

DOC. SRCE	DOC. DATE	NUMBER	REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
GE	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-PAYLOAD IMPACT V2A	15JE72	HB	002
NA	25FE72	FM347234		PAN AMERICAN APPROACH TO SHUTTLE CREW SEL/TRNG/ASSIGN	15JE72	HB	003
NA	15MA72	MSC-04217	C	SPACE SHUTTLE GN+C DESIGN EQUATION DOCUMENT	15JE72	HB	004
GR	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-MASS PROPERTIES V3	15JE72	HB	005
GE	15MR72	MSC-03824		SS PHASE B EXTENSION FINAL REPORT-EXECUTIVE SUMMARYV1	15JE72	HB	006
MD	15MR72	MDC-E0558		TECHNICAL REPORT SYSTEM + ORBITER PART 2 VOL 1	15JE72	HB	007
MD	15MR72	MDC-E0558		TECHNICAL REPORT SYSTEM + BOOSTER PART 2 VOL 2	15JE72	HB	008
MD	15MR72	MDC-E0558		FINAL MASS PROPERTIES REPORT PART 4	15JE72	HB	009
MD	15MR72	MDC-E0558		DEVELOPMENT REQUIREMENTS PART 3	15JE72	HB	010
MD	15MR72	MDC-E0558		TECHNICAL REPORT-MMC ACTIVITY PART 2 VOL 3	15JE72	HB	011
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-TECHNICAL SUM.ADD.A-BOOSTER	15JE72	HB	012
LC	15MR72	NAS826362		SPACE SHUTTLE CONCEPTS TECHNICAL REPORT VOL 2	15JE72	HB	013
MD	15MR72	MDC-E0558		EXECUTIVE SUMMARY PART 1	15JE72	HB	014
NR	15MR72	MSC-03333		SS PHASE B FINAL REPORT-MASS PROPERTIES STATUS REPORT	15JE72	HB	015
GR	15MR72	MSC-03824		SS PHASE B FINAL REPORT-TECHNICAL REPORT V2	15JE72	HB	016
NA	09FE72	EG13728		SPACE SHUTTLE GUIDANCE AND NAVIGATION REVIEW	15JE72	HB	017
BC	04JA72			STUDY OF MOTION SYSTEM REQ. FOR SIM. OF ADV. SPACECR.	15JE72	HB	018
NA	15MR72	MSC-06720		SOURCE DOCUMENTATION LIST VOL 2 CAT 2	15JE72	HB	019
NA		RFP		SPACE SHUTTLE PROGRAM REQUEST FOR PROPOSAL PHASE CD	15JE72	HB	020
NA	14JA72		A	SPACE SHUTTLE AVIONICS CONFIGURATION DEFINITION DATA	15JE72	HB	021
MM	DC71	MSC-05218		PREL. DES. OF SHUTTLE DOCKING AND CARGO HANDLING SYS.	15JE72	HB	022
NA	15MR72			DATA PKG FOR SHUTTLE TRAINING AIRCRAFT DEFINITION	15JE72	HB	023
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-EXECUTIVE SUMMARY V1	15JE72	HB	024
NR	15MR72	MSC-03332		SS PHASE B FINAL REPORT-TECHNICAL SUMMARY V2	15JE72	HB	025
NA	15NV71	MSC-03690		SS ORBITER GN+C S/W FUNCT.REQ. VERTICAL FLIGHT OPNS.	15JE72	HB	026
NA	JA70	NHB8040.2		APOLLO CONFIG.MGT. MANUAL	07JL72	HB	027
NA	01DC71	MSC-04217	B	SHUTTLE GNC DESIGN EQNS VOL.1	15JE72	HB	028
NA	01DC71	MSC-04217	B	SS GN+C DESIGN EQUATIONS-PREFLIGHT THRU ORBIT INS. V2	15JE72	HB	029
NA	01DC71	MSC-04217	B	SS GN+C DESIGN EQUATIONS-ORBITAL OPERATIONS V3	15JE72	HB	030
NA	01DC71	MSC-04217	B	SHUTTLE GNC DESIGN EQUATIONS VOL 4 DEORBITAL ATM OPNS	15JE72	HB	031
NA	15JE72		B	PROGRAM PLAN C ULASKY	15JE72	HB	032
NA	15MR72	MSC-06720		SOURCE DOCUMENTATION LIST VOL 1 CAT 1	15JE72	HB	033
NA		INDEX		SPACE SHUTTLE DATA LIST	15JE72	HB	034
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DOC. SRCE	DOC. DATE	NUMBER	REV	DOCUMENT TITLE	DATE RECD	LOCN	SEQ NO.
NA	21JF69	FHP-5-2		BLD NO 5 MISSION SIMULATION + TRAINING FACILITY	05OC72	HD	245
				AIR TRAFFIC CONTROL SYS-DIGITAL SIMULATION FACILITY	05OC72	HD	246
				A REAL TIME RADAR SIMUL USING APL DIGITAL COMP LINKS	05OC72	HD	247
				A COMPARISON OF 3 TYPES OF MANUAL CONTROLS-3RD ORDER	05OC72	HD	248
IN	FE72			INERTIAL SYS FUNCTIONAL + DESIGN REQ FOR ORBITER-B	05OC72	HD	249
				REAL TIME SPACE VEHICLE + GSE SOFTWARE SIMULATOR-COO	05OC72	HD	250
				TORQUE SENSITIVITY A-FUNCTION OF KNOB RADIUS + LOAD	05OC72	D	251
MI	AG72	E-2708		ENERGY MANAGEMENT DURING SHUTTLE TRANSITION	05OC72	HD	252
	AF72			SYSTEM 86 LIBRARY	05OC72	HD	253
NA	JI72	MSC-07036		SIMULATION OF MANNED SPACE FLIGHT FOR CREW TRAINING	06OC72	HD	254
SI	27SP72			PROGRAMMING ELEMENTS.SMS TRAINING-BURKE/VANBUCKEL/GRA	05OC72	HD	255
MI	MR72			PROJECT INTREX	29SP72	HD	256
IB				AP 101	10OC72		257
				VISUAL STUDY MCDONNELL-DOUGLAS	10OC72		258
				COMPUTER STUDY MCDONNELL-DOUGLAS	10OC72		259
				SHUTTLE DEVELOPMENT SCHEDULE	10OC72		260
				EMS CREW TRAINING SCHEDULE	10OC72		261
GE	AF72	NAS911946		REMOTE MANIPULATOR DYNAMIC SIMULATION	10OC72		262
MA	AF72	CR123570		SHUTTLE PROP SYS ONBOARD CHECKOUT + MONITORING DEV	10OC72		263
CC	06MA72	CR61376		COMPUTER SYS SIMULATION + ANALYSIS	10OC72		264
IN	16FE72	N7221203		ADV SFTWR TECH FOR DATA MGMT SYS VOL 3-PROG LANGUAGE	10OC72		265
IN	16FE72	N7221205		ADV SFTWR TECH FOR DATA MGMT SYS VOL 2-SSV EXEC SYS	10OC72		266
IN	16JF71	N7221204		ADV SFTWR TECH FOR DATA MGMT SYS VOL 1-SSV SW STUDY	10OC72		267
MS	07MA72	CR123569		FLIGHT PROGRAM LANGUAGE REQMT	10OC72		268
SF	- 71			ATC FOR THE SEVENTIES PART 1	10OC72		269
SF	- 71			ATC FOR THE SEVENTIES PART 2	10OC72		270
SF	- 71			ATC FOR THE SEVENTIES PART 3	10OC72		271
AI	08MA71	A71-30381		PROG TECH FOR REAL TIME SYS FORM EXPERIENCE W EXPERMT	10OC72		272
NA	DE70	TN-D-6088		DESIGN E OPER CHAR OF CENTRAL ONLINE DATA PROC-LANGLY	10OC72		273
TN	AF71	TSC-FAA71		CONCEPTUAL NETWORK MODEL OF AIR TRANS SYS	10OC72		274
MA	JA71	CR118310		VIRTUAL MEMORY SYS DESIGN	10OC72		275
RA	JA71	CR118868		EXPERIENCE WITH EXTENDABLE COMPUTER SYS SIMULATOR	10OC72		276
MT	MR71	MTR-1995		AIRCRAFT MOCKUP COMPUTER PROGRAM SPECIFICATIONS	10OC72		277
AI	05MR68			NOMOGRAM FOR AIRCRAFT RUDDER PEAL DESIGN	10OC72		278
AI	05MR69			OPTIMUM KNOB DIAMETER	10OC72		279
AI	05MR69			DESIREABLE DIMENSIONS FOR CONCENTRIC CONTROLS	10OC72		280
MI	JF69	CR106370		MANUAL CONTROL OF UNSTABLE VEHICLES-KINESTHETIC CUES	10OC72		281
RA	AU69	RM-6027		ONLINE DEBUGGER FOR G5560 ASSEMBLY LANGUAGE PROGRAMS	10OC72		282
MA	JA70	CR110445		METHOD FOR UNIFIED HARDWARE-SOFTWARE DESIGN	10OC72		283

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2.0 Mission Envelopes

REF. KEY

2.1 Space Missions

This discussion breaks the full space missions down into three phases-launch to orbit insertion, orbit through the de-orbit maneuver to the entry point (.05 g or h = 400,000 ft.) and finally entry to touchdown and roll-out. Although the potential for any of these phases is unlimited for the space shuttle as presently conceived with expansion capabilities, the study to determine simulation requirements can be approached under these three phases of the space mission and then investigation of the requirements under specific mission or tank/maneuver dictates to determine the total requirements. The latter is addressed here for the Space Mission/Envelope under Payloads, specific Reference Mission, Timelines, Aborts, and Mission Operations.

2.1.1 Launch Phase

REF. KEY

2.1.1.1 Requirements

Shuttle Space missions will be launched from both KSC and the WTR. Crew ingress is expected about one hour prior to launch.
1 Prelaunch activities are required for checklist and countdown activity including:

- Systems activation
- Powered flight navigation initialization
- Launch control monitor initialization
- Timing functions
- Checklist
- Transfer from External to Internal power

2 3 The shuttle orbiter will be launched with an external HO tank attached
4 to the lower fuselage, and with two recoverable 156 inch solid rocket motors attached to the External Tank. Thrust during the first stage
2 boost will be provided by the Solid Rocket Motors (SRM) and the Orbiter Main Engines (ME) firing in parallel. The vehicle will be controlled by Orbiter main engine TVC, augmented during periods of high dynamic pressure by orbiter elevons and rudder (chiefly for roll control). Guidance will command a vertical rise for five seconds after liftoff, followed by a pitch profile. Roll to flight azimuth is accomplished during pitchover. Attitude commands are programmed as functions of velocity, to provide good orbiter engine-out performance. The pitch profile is designed to minimize gimbal requirements. Body-mounted accelerometer feedback is included in the control system from about 25 seconds after launch to about 95 seconds after launch for load relief. At staging, the

6 SRM's will be separated from the External Tank/Orbiter and the Orbiter ME's will continue firing to orbit insertion. After staging, an adaptive guidance program is initiated for steering to orbit insertion, Orbiter main engines will be throttled to limit longitudinal acceler-
7 ation to a maximum of 3 g's. After insertion, the external tank will be vented, then jettisoned and deorbited. Typical launch envelope parameters are:

		Easterly	Resupply	Polar
8 Max q:	$q = lb/ft^2$	639	642	648
	t = sec			54
	h = ft			35,300
	V_r = ft/sec			1,760
	weight lb.			3,137,564
	Mach			1.81
	Thrust (lb)	7,982,000	7,982,000	7,982,000
Staging:	α Relative	34.2	34.3	34.2
	$q = lb/ft^2$	67.8	66.1	63.5
	t = sec	109.1	109.1	109.1
	h = ft.	130,300	131,200	132,800
	V_r = ft/sec	4,035	4,062	4,125
	weight (lb.)	2,035,125	2,019,125	1,994,125
	Mach	3.68	3.70	3.76
	Thrust (lb.)	1,410,000	1,410,000	1,410,000
Orbit Insertion:	q =	0	0	0
	t = sec	546.8	548.4	551.3
	h = N.M.	50	50	50
	downrange (N.M.)	792.2	808.3	840.2

9 The Orbiter/External Tank will insert into a 50 X 100 N.M. altitude orbit.

2.1.1.2 Rationale for Assumptions

Not applicable

REF. KEY

2.1.1.3 Data References

1. 10 pages 6 - 24
2. 20 page I-1; 166, pp. 2 - 34, 2 - 35, 2 - 37
3. 8 page C. 1-4
4. 16 pages 5 - 20
5. 12 pages 3 - 67
6. 20 page IV-6; 166, pp. 2 - 34, 2 - 46, 3 - 60
7. 13 pages 3 - 5
8. 166 page 2 - 17
9. 20 page IV-5

2.1.2 Orbit and Deorbit Phases

2.1.2.1 Requirements

Any of a variety of orbital operations may be required depending upon the objectives of a given mission. Among missions currently foreseen are:

1 Satellite Placement

Satellite Retrieval

Orbital Observation

Orbit-to-Orbit Shuttle Satellite Deployment and Retrieval

Propellant Delivery

2 Propulsive Stage Delivery

Space Station Modular Build-up

REF. KEY

- Space Station Resupply
- Rescue

and various combinations of the above. Burns to alter the orbit will
3 be performed by either the Orbital Maneuvering System (OMS) or the
Reaction Control System (RCS). The orbiter will be able to attain
4 (considering payload and propellant loading) and return from orbits
as high as 500 X 500 N.M. Rendezvous and Docking capability will be
5 provided. On-board rendezvous capability with cooperative targets
displaced less than 300 N.M. will exist. The basic mission length
will be 7 days but the vehicle design will not prevent orbital mission
durations of up to 30 days. Orbital inclinations range to over 90 degrees.
Deorbit will be accomplished using the OMS.

2.1.2.2 Rationale for Assumptions

Not applicable.

2.1.2.3 Data Referneces

1. 46 pages 3-2, 3-3
2. 47 pages 4-19, 4-20, 4-21
3. 20 pages IV-5, IV-19, 20
4. 20 page IV-13
5. 20 page IV-9
6. 20 page IV-20

2.1.3 Entry and Landing Phases

2.1.3.1 Requirements

The orbiter will be able to re-enter and land safely from
1 orbits of up to 500 X 500 N.M., with payloads of as much as 40,000 pounds.
2 Entry acceleration loads will not exceed 3 G's. The orbiter will have a
crossrange capability of about 1100 N.M. (adequate to return to
3 launch site after one revolution) to be attained during entry (between
4 400,000 feet altitude and 50,000 feet altitude), and downrange of about
5000 N.M.

The entry and landing phase can be separated into operational modes:

5 Re-entry: Energy management versus temperature, g-load and skip-out constraints

Transition: Maneuver from back side to front side of L/D curve

Terminal Area: Transition to final approach

Landing: ILS

6 Typical entry trajectory parameters for a high crossrange entry are:

400,000 feet	inertial velocity	25,612 ft/sec
	inertial flight path angle	-0.91°
	Δt to 50k feet	33 min.

7 Depending on mission payload requirements, etc., airbreathing engines
may be available for use during the final phase. If included, they
8 will permit 15 minutes loiter time at 10,000 feet altitude (standard)
9 day). The orbiter approach speed will be about 250 kts and landing
speed about 152 kts. Touchdown angle of attach is estimated at 13°.
10 Sink rate at touchdown will not exceed 10 ft/sec. The orbiter will be
11 able to land on a 10,000 foot long, 150 foot wide wet runway.(sea level)
on a 103° F day.

2.1.3.2 Rationale for Assumptions

Not applicable

2.1.3.3 Data References

1. 20 pages IV-13, IV-18
2. 20 page IV-6
3. 18 page IV-5
4. 16 pages 2 - 4
5. 46 pages 3-7, 3-8; 7 pages B.2-12, B.2-14
6. 166 page 2 - 77, 2 - 61

Ref.
Key

7. 20 page IV-9
8. 20 page IV-7
9. 7 page B.2-11; 37 pages 4-27, 5-36; 166 Pp. 2-77, 2-52
10. 20 page IV-17
11. 20 page IV-10

2.1.4 Payloads, Deploy and Retrieval

2.1.4.1 Requirements

1 The shuttle orbiter will be able to transport payloads of up to 65,000 pounds weight (depending on mission). The shuttle payload bay
2 will have a clear volume 60 feet long and 15 feet in diameter. Payloads, may reach 15-foot diameters and 50-foot lengths, including attachment fittings and handling rings. The vehicle will also provide for up to six
3 additional crewmen (for short durations). The vehicle will provide electrical
4 power and data transmission connections with the payload while attached to
5 the vehicle. Multiple (up to five) payloads and payloads of less than 15
6 feet diameter can be accomodated using adapter mechanisms. The orbiter
5 will be configured to allow crew access (pressure suit) to the payload bay in flight. Payload center-of-mass locations will be permitted some variation;
7 ranging from a 10-foot envelope for maximum payload to a 60-foot envelope for very small payloads. Among payloads currently forseen are the following:

8 o satellites
o propulsive stages
o supply modules
3 o additional crew
9 o space station build-up modules

A Payload manipulation will be accomplished by two manipulator arms, roughly similar to human arms.

2.1.4.2 Rationale for Assumptions

5 A. The RFP merely specifies a payload deployment and retrieval

mechanism. However, contractors discussing the subject appear to largely agree on such a humanoid device.

2.1.4.3 Data References

1.	18	page IV-5
2.	18	page IV-6
3.	18	page IV-9
4.	18	page IV-40
5.	18	page IV-8
6.	33	page 3-19
7.	19	page IV-39
8.	46	pages 3-2 through 3.4
9.	33	page 4-19
10.	34	page 6-117
11.	5c	page B.3-KO
12.	20	Section II

REF. KEY

2.1.5 Reference Missions

The missions for the shuttle vehicle consist of some combination of eight operational sequences. These are:

Ascent

Staging

Tank Disposal

On Orbit

Return to Earth

Rendezvous

Payload Handling

Abort

All missions are planned to include the first five. The missions may include all or some combination of the remaining sequences. These missions are limited in the discussion to:

Vertical Tests

Easterly

Resupply

South Polar

which are representative of the missions to be flown in the space shuttle.

REF. KEY

2.1.5.1 VERTICAL TESTS

2.1.5.1.1 Requirements

10 A series of vertical test flights will be made to approach the critical tests of the system and subsystems in increments for reasons of safety. The flights will carry cargo payloads and/or Development Flight Instrumentation (DFI). The tests will also evaluate procedures and the requirements for the Space Tracking and Data Network (STDN). The tests will demonstrate the ability for payload placement and retrieval, and the ability to rendezvous, dock and transfer cargo. The missions will test:

10
- Structure
- Aerodynamics
- Thermodynamics
- OMS & ACPS
- Navigation
- Communications
- ECLS
- Effects of cargo center of gravity

178 The DFI is expected to require approximately 2,803 measurements for performance analysis as follows:

	Development Flight Instrumentation											
	Meas Quantities									** Meas Use		
Subsystem	Pressure	Thermal	Dynamic	Electrical	Event	Position	Load	Miscellaneous	Total	PCM TM/Rec	Wideband Rec	Wideband TM
Aero-surfaces	195	253	–	–	–	–	–	–	448	448	–	–
Vehicle struct	–	72	286	–	–	–	198	–	556	270	286	57
Thermal prot	20	831	–	–	–	–	–	–	851	851	–	–
MPS	19	11	–	–	22	–	–	–	52	52	–	–
OMS	–	7	–	–	–	–	–	–	7	7	–	–
RCS	–	12	39	–	–	–	–	–	51	12	39	–
ABPS	80	80	–	–	–	–	–	–	160	160	–	–
GN&C	24	6	–	–	172	10	–	–	212	212	–	–
COMM	–	–	–	–	26	–	–	–	26	26	–	–
EPD	–	10	–	9	–	–	–	–	19	19	–	–
Hydraulic pwr	17	32	–	–	–	6	–	–	55	55	–	–
ECLSS	–	–	–	–	–	–	–	–	–	–	–	–
Flt crew supt	–	18	5	–	–	–	4	6	33	30	3	–
Instrumentation	–	25	–	12	3	–	–	–	40	40	–	–
Elect power gen	23	50	18	–	–	–	–	–	91	73	18	–
Mech power gen	–	12	6	–	–	–	–	–	18	12	6	–
D&C	–	–	–	–	–	–	–	–	–	–	–	–
Orbiter totals	378	1419	354	21	223	16	202	6	2619	2267	352	57
SRM totals	–	18	12	–	–	–	–	–	30	–	–	–
ETS totals	2	112	30	–	10	–	–	–	154	–	30	–
Grand totals	380	1549	396	21	233	16	202	6	2803	2267	382	57

ORBITER DFI (VERTICAL) MEASUREMENT SUMMARY

REF. KEY

Constraints for the first mission(s) are:

- Flight profile will not subject vehicle to greater than the design loads, heating and dynamic pressure.
- 10 Range safety restrictions.
- STDN or similar communications, data and tracking required for:
  - Insertion
  - Booster entry and recovery
  - Early revolutions of orbit
  - Orbiter entry and recovery
- Daylight VFR weather at launch and Primary/ Secondary landing sites.
- Orbiter to carry JP fuel for 100 miles cruise.
- "Deadstick" landing optimized entry.

16 178 54 The first vertical test flight will be unmanned with a fully operational orbiter. In this case, the mission would be limited to:

- Once-around or
- Five orbits maximum

16 Control backup to the fully automatic systems would be either from the Chase Aircraft or from the ground. Extensive training will be required in either case.

54 All deorbit cases range through KSC, Edwards AFB, Hawaii and Guam as potential landing sites for a

REF. KEY

20-orbit mission with KSC as prime. If the orbit is circularized at 138 N.M. altitude, the deorbit ΔV

54 is 270 fps. If circularization is accomplished, the orbit altitude is expected to be in the range 100 to

178 350 N.M. Better STDN coverage is achieved at higher altitudes while less contingency RCS deorbit propellant is required at the lower altitudes. The nominal orbit

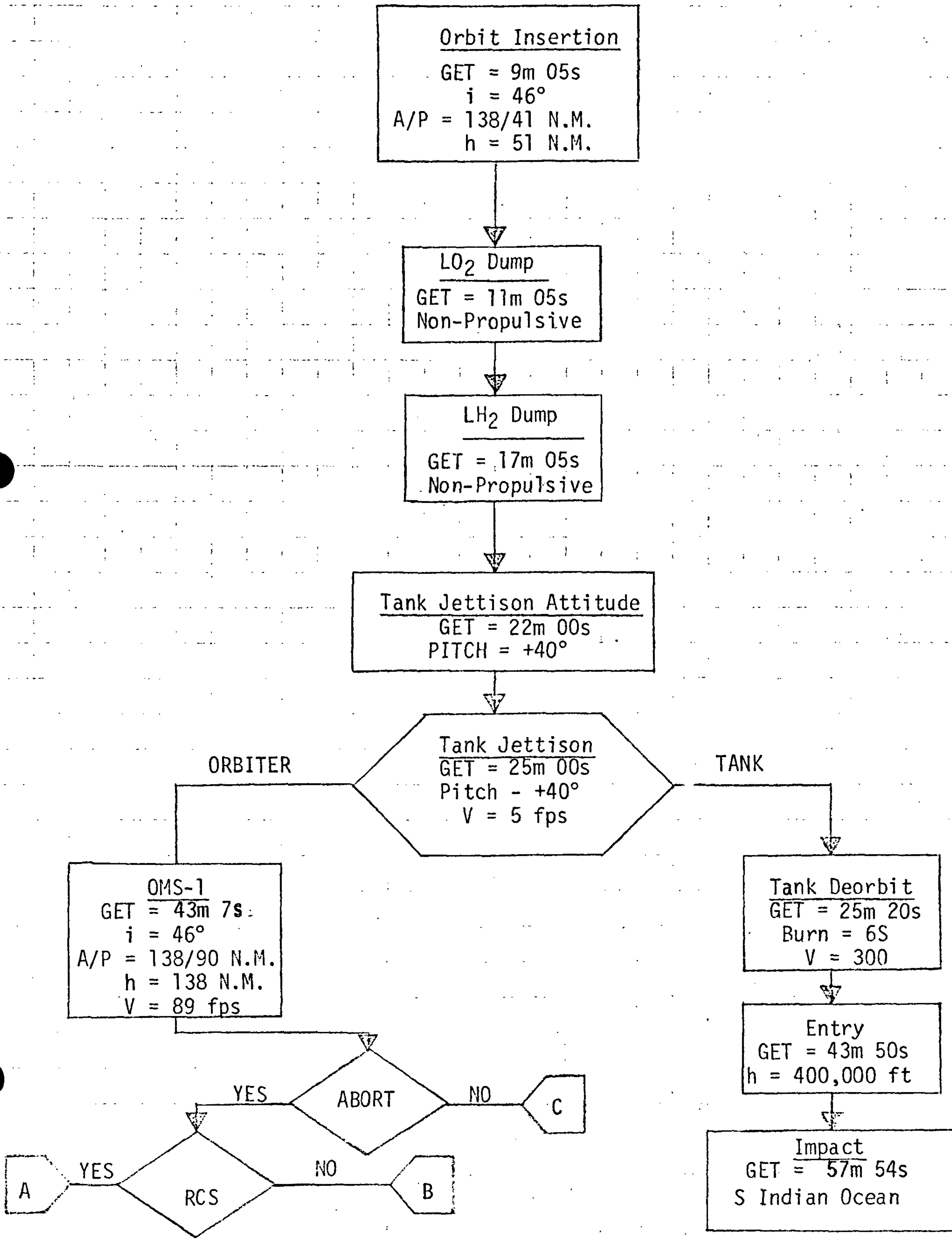
178 inclination is 28.5°. The higher inclinations (40°) provide best landing site accessibility while the lower inclinations provide the best STDN coverage.

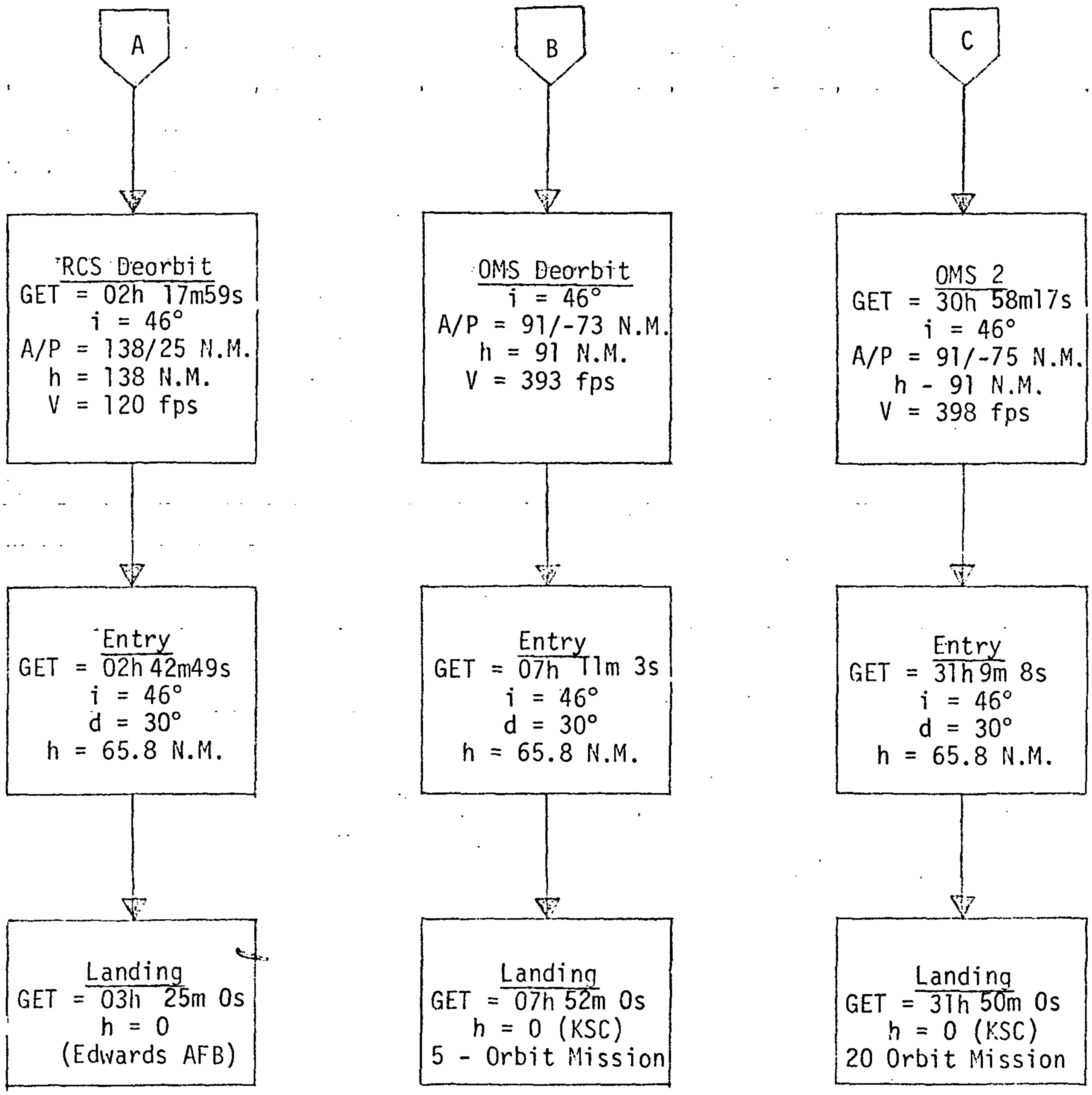
2.1.5.1.1.1 STDN

The Spacecraft Tracking and Data Network (STDN) is

54 not yet defined, however the assumed number is 22 located in a range from 35.401596S to 57.60 N geocentric latitudes. Orbital inclinations of 25° to 42° afford the best coverage with 32° being optimum.

2.1.5.1.1.2 EVENT TIMELINE




A
B
C
RCS Deorbit
GET = 02h 17m59s
i = 46°
A/P = 138/25 N.M.
h = 138 N.M.
V = 120 fps
OMS Deorbit
i = 46°
A/P = 91/-73 N.M.
h = 91 N.M.
V = 393 fps
OMS 2
GET = 30h 58m17s
i = 46°
A/P = 91/-75 N.M.
h - 91 N.M.
V = 398 fps
Entry
GET = 02h 42m49s
i = 46°
d = 30°
h = 65.8 N.M.
Entry
GET = 07h 11m 3s
i = 46°
d = 30°
h = 65.8 N.M.
Entry
GET = 31h 9m 8s
i = 46°
d = 30°
h = 65.8 N.M.
Landing
GET = 03h 25m 0s
h = 0
(Edwards AFB)
Landing
GET = 07h 52m 0s
h = 0 (KSC)
5 - Orbit Mission
Landing
GET = 31h 50m 0s
h = 0 (KSC)
20 Orbit Mission

REF. KEY

2.1.5.1.2 Assumptions

During launch-abort the various launch abort modes capability must be simulated with automatic as well as manual take-over. If unmanned, a simulation for remote control of the fly-back vehicle must be provided with

A requirements yet to be defined. A principle subject of orbital flight testing will be the use of RCS fuel and various failure conditions for the OMS. Accurate state vectors must be maintained to test fly-back with various cross-ranges to the selected landing sites and alternates. Control will be a **major** consideration during entry - especially during the hypersonic-supersonic transition *followed by energy management and guidance to the landing*

B site. Document 54 assumed a different booster than the NAR Proposal (document 166). To attempt to correct for this, insertion time was adjusted to approximately match typical NAR proposal trajectories. This required adjustment of other times as well. LO_2 and LH_2 dump times were assumed to be functions of insertion time only, and were adjusted likewise, other times were assumed functions of geographic position, and were adjusted accordingly, using information in 166, p. 2-17 and 54, p. 23 to estimate the required bias.

2.1.5.1.4 References

10

16

54

REF. KEY

2.1.5.2 Easterly Launch

2.1.5.2.1 Requirements

The Easterly Launch mission is baselined to a 100 n.m. circular orbit. The mission will be launched due east and requires payload delivery capability of
20 PP.IV-5 65,000 pounds with the orbiter ABES removed. The purpose of the mission is assumed to be placement and/or retrieval of a satellite. The orbiter on-orbit ΔV requirement is 950 f.p.s. from the OMS and 120 f.p.s. from the RCS. Approximately 60% of
36 PP. 4-23 the projected missions are due east launches. The missions involve rendezvous for satellite deploy and/or retrieval, Space Station support, Space Tug (OOS) deploy/retrieval and service or maintenance of space vehicles. The mission may require waiting in orbit for OOS missions which is comparable to station-keeping in Space Station support missions. The mission operations will include:

46 PP. 3-12

- Navigation
- Guidance
- Control
- Communications
- Time Schedules
- Mission/Subsystem Constraints Monitoring
- Subsystem Monitoring and Operations
- Planning

2.1.5.2.2 Timelines

REF. KEY

LAUNCH

REF. KEY	EVENT	GET SEC.	ALTITUDE FT.	VELOCITY FPS	REL.GAMMA DEG.	q PSF
7						
8	Max Q	54	36000		60	639
	SRM Cut-Off	109.1	130300	4035	35	67.8
	SRM Impact	439	0	-	-	-
	Orbit Insertion		303980	24545	0	-

ORBIT

REF. KEY	EVENT	GET d h m s	APOGEE N.M.	PERIGEE N.M.	Δ V FPS	PURPOSE
	Insertion	00h09m07s	100	50	0	Base
	Tank Jettison	00h22m00s	100	50	3.0	Tk. Separation
	OMS-1	00h50m35s	117	100	121.7	Hohmann
36	OMS-2	01h34m48s	117	117	30.6	Circularization
	Release Payload	03h00m00s	117	117	-	Payload
	OMS-3	5d15h00m00s	141	117	42.8	Hohmann
	OMS-4	5d15h54m41s	160	135	127.0	Hohmann
	OMS-5	5d16h29m44s	160	160	47.0	Circularization
	TPI V	5d18h26m48s	171	160	21.0	Rendezvous
	TPF Maneuvers	5d18h59m33s	170	170	28.0	Rendezvous
	Retrieve Payload	5d20h00m00s	170	170	-	Payload
	Deorbit	6d20h16m00s	170	TBD	250.0	Deorbit

ENTRY

EVENT	GET d h m s	ALTITUDE FT.	VELOCITY FPS	DEG.	q
Entry	6d20h36m20s	400,000	25,780	-1.1	
Landing	6d21h36m20s	0	-	-	-

2.1.5.2.3 Assumptions

None

2.1.5.2.4 References

20 pp. IV-5

36 pp. 4-23

46 pp. 3-12

8 Figure 6.2-1

35 pp. 4-24 through 4-30

166 p. 2 - 17

REF. KEY

2.1.5.3 Resupply

2.1.5.3.1 Requirements

20 pp IV-5 The baseline mission is resupply of a station in a 270/270 n.m. orbit at 55° inclination. Launch is from KSC to accomplish rendezvous using a 17 orbit co-elliptic rendezvous sequence. The payload is 25,000 pounds. The ABES will be installed with fuel to provide 15 minutes loiter at 10,000 feet altitude prior to landing. To meet the latter requirement, the payload may be reduced. (20 IV-20) The orbiter on-orbit translational Δv requirement is 1400 ft/sec by the OMS, and 120 ft/sec from the RCS. Initial orbit insertion will be into a 50 by 100 n.m. orbit at 55° inclination. The resupply operations will include docking, partial crew transfer, equipment transfer, (32 Table 3-1) undocking and separation, station keeping for 28 days with experiment operations and re-rendezvous for crew retrieval. Pointing accuracy and stability is required for Astronomy Experiments and maximum limits for the ECLS system.

2.1.5.3.2 Timelines

REF. KEY	EVENT	GET hms	Δv F.P.S.	Apogee/Perigee N.M.	Altitude N.M.
8	INSERTION	00:09:08	---	100/5	50
	PHASING	00:49:50	130	123/100	100
	HEIGHT	22:14:58	282	260/123	123
	COELLIPTIC	23:00:47	238	260/260	260
50	TPI	24:30:54	22	272/260	260
	BRAKING	25:05:04	45	270/270	270
	DOCKING	25:35:04	10	270/270	270
	SEPARATION	67:20:05	10	270/265	270
	DEORBIT	70:20:05	494	270/-10	270
	ENTRY	70:52:05	---	---	65.8
	LANDING (KSC)	71:32:05	---	---	0

REF. KEY

NOTE: This timeline is for a minimum stay after resupply.

2.1.5.3.3 Assumptions

None.

2.1.5.3.4 References

20

32

166

2.1.5.4 SOUTH POLAR

2.1.5.4.1 REQUIREMENTS

The South Polar mission will be launched from the WTR for payload delivery or retrieval in a 100 n.m. circular orbit and return to the launch site after two revolutions. Rendezvous and docking operations will be completed within one revolution. The payload required is 40,000 pounds with the ABES removed. The orbiter translational required is 500 ft/sec from the OMS and 150 ft/sec from the RCS. The mission requires a day or night launch, rapid rendezvous, payload deploy or retrieval, deorbit and entry to the landing site.

2.1.5.4.2 TIMELINES

BOOST

EVENT	GET SEC	INERTIAL VELOCITY FT/SEC.	ALTITUDE FT.	DOWN-RANGE n.m.
Staging	109.1	4,125	132,800	
Orbit Insertion	551.3	25,945	303,808	840.2

EVENT	WEIGHT LBS.	CG-INCHES X	Y	Z	MOM. of INERTIA-SLUG-FT^{2}X10^6 IXX	IYY	IZZ
Lift-off	4,656,739	1558.9	.4	446.0	40.92	373.80	402.06
Max Q	3,137,564	1453.9	.6	441.3	22.00	259.92	271.81
SRM Burn-Out	1,994,125	1309.8	.9	441.5	9.22	166.53	167.42
Post Staging	1,658,425	1176.0	1.1	440.1	4.68	112.39	109.14
Orbit Insertion	310,034	k933.6	5.7	613.9	2.18	20.10	19.37

(CG in Integrated vehicle reference system; tank nose at X=200 in. and tank C_L at Z=400 in.

KEY REF.

2.1.5.4.2 Timelines

Entry Timeline
i = 90°

GET (seconds)	0	392	900	1,185	1,410	1,710	1,855	1,975	2,575
VELOCITY (ft/sec.)	25,612	25,218	20,150	13,600	8,000	3,260	1,550	910	300
GAMMA (degrees)	-.91	-.11	-0.35	-0.45	-0.66	-3.00	-7.00	-8.20	0
DYNAMIC PRESSURE (16/ft^2)	0	24	63	106	81.0	172	150	150	105
DOWN-RANGE (n.mi.)	0	1,500	3,500	4,000	4,500	4,700	4,800	4,800	4,800
CROSS-RANGE (n.mi.)	0	0	100	250	500	800	1,000	1,085	1,085
ANGLE OF ATTACK (degrees)	34.0	34.0	34.0	34.0	34.0	16.0	10.0	10.0	15.0
BANK ANGLE (degrees)	30	30-75	55.0	50.0	15.0	15.0	0.0	0	0
LOAD FACTOR (g's)	0	0.31	0.82	1.40	1.07	1.08	0.99	--	--
ALTITUDE (ft)	400,000	248,500	214,000	179,000	158,000	100,600	72,300	50,000	0

166 pp 2-77

2.1.5.4.3 Assumptions

Although information is not available, it is assumed that the launch will be targeted to intercept any payload to be retrieved such that the insertion is coplanar and range to the target is short (a few miles at most) in order to accomplish the rendezvous within one orbit. Time would not permit many additional experiments. The crew work load would be maximum with no rest periods. There is some ambiguity between the reference documents. It is expected that the launch-entry will require most of one orbit and that the intent is for rendezvous to be accomplished within one orbit making the total mission equal to two revolutions. This is in agreement with reference 32, but disagrees with reference 20.

2.1.5.4.4 References

15

20

32

16

166

2.1.6 Timelines

2.1.6.1 Requirements

REF. KEY	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	PRELAUNCH	
	• Transfer Vehicle to Pad	-00:34:00:00
	• Connect Interfaces	-00:27:00:00
10	• Interface Tests	-00:26:00:00
pp 6-23,24	• Vehicle Power Checks	-00:25:00:00
	• Pressurize Vehicle Pneumatics to 1500 psi	-00:23:00:00
57	• Set up	-00:23:00:00
pp 1.1-10	- Pad/LUT Water/FIREX Systems	
	- ECS	
	• Functional Tests	-00:14:00:00
	- Propulsion	
10	- Hydraulics	
pp 6-23	- Avionics	
	• Integrated Systems Tests	-00:13:00:00
	• Activate Fuel Cells	-00:08:00:00
	• Fuel Cells to Standby	-00:07:00:00
57	• Secure Doors for Flight	
pp 1.1-11	• Clear Pad	
	• IMU Alignment	
10	• Standby Status	-00:02:30:00
pp 6-24	• LH2 & LOX Facility & Transfer Line Childown	-00:02:10:00
	• Orbiter Engines Chilldown	
	• LH2 & LOX Fill and Maintain Replenishment	-00:01:54:00
57	• Crew Ingress Preparations	-00:01:15:00
pp 1.1-12	• Crew Entry	-00:01:07:00

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	PRELAUNCH - Continued	
	Fuel Cell Cryo Heaters on (If Required)	
	Cabin Doors Secured	-00:00:55:00
	Functional Checks	-00:00:50:00
	Final Guidance Update	-00:00:20:00
	Start APU's	-00:00:09:00
	Range Tests	
	Fuel Cells on DC Bus	
10 pp 6-24	Pressurize Tanks	-00:00:05:00
	Power to Internal	
	Internal Checks	
	Main Engines Start	-00:00:00:03.9
	Unlock ME TVC Activators	-00:00:00:00.3
	SRM Ignition	-00:00:00:00.2
	Tip-Lock Release	-00:00:00:00.1
	LIFTOFF	00:00:00:00
46 pp 3-14	Powered Flight Navigation	
	Guidance and Control	
21 p 292	Targeting	
	Heads Down	

KEY REF.	CMD PILOT	PILOT	MISSION SPC.	PAYLOAD SPC.
10 pp 6-35	Ascent Monitor	Nav. Monitor	Subsys. Monitor	Payload Monitor
	Pilot Backup	Comm.	Failure Monitor	
	Abort Monitor	CMD Pilot B/U	Fault Isolation	
26			CB Control	
[illegible]/			FC Control	
pp 1.2-11			ECS Control	

KEY REF	EVENT/ACTIVity KSC LAUNCH-EASTERLY	GET d h m s
	LIFTOFF (Continued)	
	Ground Functions:	
	• Monitor Boost Trajectory	
10	• Communications	
pp 6-35	• Provide Procedures, Computational, Flight Dynamics	
	• Predict Booster Impact Point	
8	• Abort Mode is 0	00:00:00:05
166	• Start Pitch/Roll Manuever	00:00:00:30
pp 2-15 pp 2-35	• ASRM Jettison	
	MAX Q	00:00:00:54
	• Communications	
	• Systems Monitoring	
	• Abort Monitoring	
	• G, N&C Monitoring	
	• Motion, Audio Cues	
	• Caution and Warning Monitoring	
8	• Abort Mode is 1	
	• Aero + TVC Control	
	SRM BURN-OUT	00:00:01:46.3
	• Staging Sequence Monitor	
	• Engines Performance Monitor	
	• Trajectory Monitor	
	• Subsystems Monitor	
	• Payload Monitor	
	• Abort Monitor	
	• Audio/Motion Cues	
	• Abort Mode is 2	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	STAGING	00:00:01:49.3
	• Staging Sequence	
	• Powered Flight Navigation	
	• Insertion Phase Guidance	
	• Subsystems Status	
	• Trajectory Monitor	
	• Abort Monitor	
	• Communications	
	• Audio/Motion/Visual Cues	
8	• Abort Mode is 2, then 3	
	ORBIT INSERTION	00:00:09:06.8
	• Main Engine Post-Burn Procedures	
	• Orbital Parameters	
	• Communications	
	• Powered Flight Navigation	
	• Systems Checkout	
	• Attitude Control	
	• Audio/Visual Cues	
8	• Abort Mode is 4	

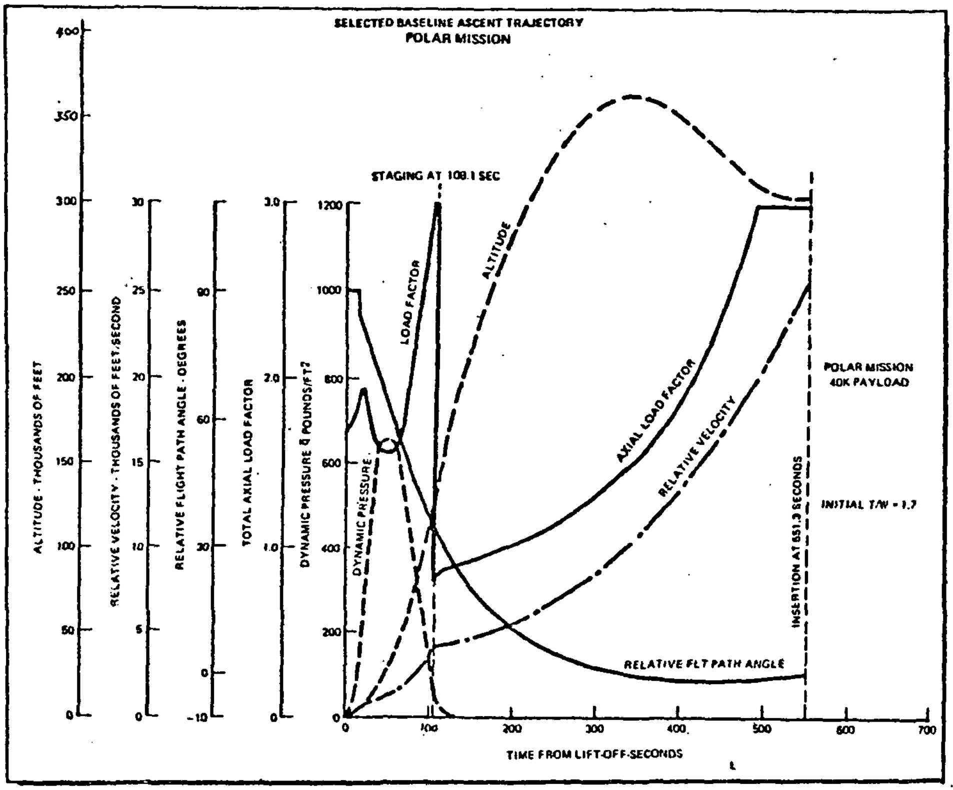
KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
167 pp 1-4		

Figure 1.2-1. Boost Trajectory Data for 156-Inch Parallel-Burn SRM Baseline Launch Configuration (-0007D Orbiter)

LAUNCH-BOOST TRAJECTORY SUMMARY

KEY REF.	EVENT/ACTIVITY	GET
	EXTERNAL TANK	
46	• LO_2 Dump	00:00:09:26
36	• LH_2 Dump	00:00:15:26
	• Attitude Maneuver and Maintenance	00:00:20:23
166 pp 2-9	- Pitch = $+40^o$	
	- db = $\pm 0.5^o/3$ axes	
	TANK JETTISON	00:00:33:30
46	• Tank ΔV in - Z = 3 f.p.s.	00:00:23:13
36 166	• Audio/Visual Cues	

F-398-8-A

REF. KEY	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
•	OPEN CARGO BAY DOORS	
46 pp 3-15	• Deploy Radiators	
	SYSTEMS AND PAYLOAD STATUS CHECKS	
	SHUT DOWN APU	
	DEPLOY ANTENNA	
	TANK LOS AND DEORBIT	00:00:33:47
36 pp 4-25	• Maintain LOS on Tank	
	- db = $\pm 10^{o}/3$ axes	
	• Deorbit Tank	
	- Deorbit ΔV = 300 f.p.s. retrograde	
	- Tank destructs in atmosphere over water	
	• Communications	
	• Visual Cues	
	IRU ALIGNMENT	
	- OBC Automatic with Manual Backup	
	BURN ATTITUDE AND HOLD	00:00:40:58
	- Horizontal In-plane, posigrade, heads-up	
	- Attitude db = $\pm 1^{o}$ then $0.5^{o}/3$ axis	
	- Initiate Powered Flight Navigation	
36 pp 4-25	- Pre-Burn Checklist	
	- Align IRU	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	HOHMANN BURN # 1	00:00:51:58
46	- OMS Burn Checklist	
	-ΔV = 121.7 f.p.s.	
	- Apogee/Perigee = 117/100	
	- Terminate Powered Flight Navigation	
	- Post Burn Procedures	
21	- Communications	
pp 292-29g	- Audio Cue	
	INSERTION	00:00:52:39
	• Heads-up	
	• Attitude db = $\pm 10^{o}$/3 axis	
	BURN ATTITUDE AND HOLD	00:01:25:00
	HOHMANN BURN # 2	00:01:36:01
36	• Align IRU	
pp 4-26	• OMS Burn Checklist	
	• Powered Flight Navigation	
	• ΔV = 30.6 f.p.s.	
	• Apogee/Perigee = 117/117	
	• Post-Burn Procedures	
	• Communications	
	• Audio Cue	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	PAYLOAD PRE-DEPLOY CHECKOUT	00:01:38:23
	• Attitude db = $\pm 45^{o}/3$ axis	
36 pp 4-36	• Activate Manipulator Arms	
	• IRU Alignment	
	• Audio/Visual Cues	
	PAYLOAD DEPLOY	00:02:26:23
	• Attitude db = $\pm 10^{o}/3$ axis	
	• Deploy Attitude	
46 p 3-16	• Deploy Payload from Cargo Bay	
	• Final Checkout and Payload Orientation	
36 pp 4-26	• Activate Payload Systems	
	• Check Extended payload	
	- Visual	
	- Data Link	
	• Coast to Release Point	
	- Close db to $\pm 0.5^{o}/3$ axis	
	PAYLOAD RELEASE	00:02:41:23
	• Retract Manipulator Arms	
36 pp 4-26	• Separate to safe distance	
	• Maneuver on Payload LOS	
Proposal pp 2-9	- Attitude db = $\pm 10^{o}/3$ axis	
	• Monitor Payload Functioning	
	• Maintain LOS on payload	
	• Visual Cues	

KEY REF.	EVENT/ACTIVITY KSC ALUNCH-EASTERLY	GET d h m s
	BURN ATTITUDE AND HOLD	05:14:50:23
	• IRU Alignment	
	• Attitude db	
	• Target Phasing and Rendezvous with Recoverable Satellite	
	HOHMANN BURN # 3	05:15:01:23
	• OMS	
	• Horizontal, In-Plans, Posigrade, HU	
36 pp 4-27	• Orbiter trails target 6.8^{o} In-Plane	
	• ΔV = 42.8 f.p.s.	
	• Post-Burn Procedures	
	• Powered Flight Navigation	
	• Apogee/Perigee = 141/117 N.M.	
	• Audio cue	
	BURN ATTITUDE AND HOLD	05:15:45:04
	• Attitude	
	• Powered Flight Navigation	
	• IRU Alignment	
	HOHMANN BURN #4	05:15:56:04
36	• OMS	
pp 4-27	• Corrective Combination Burn	
	• ΔV = 127.0 f.p.s.	
	• Apogee/Perigee = 160/135 N.M.	
	• Post-Burn Procedures	
	• Audio Cue	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	BURN ATTITUDE AND HOLD	05:16:20:07
	• Attitude	
	• Powered Flight Navigation	
	• IRU Alignment	
	• Horizontal, Posigrade, HU	
	HOHMANN BURN #5	05:16:31:07
	• OMS	
	• Coelliptic Circularization	
	• ΔV = 47 f.p.s.	
	• Apogee/Perigee = 160/160 N.M.	
36	• Post Burn Procedures	
PP 4-28	• Audio Cue	
	TARGET LOS ATTITUDE MANEUVER	05:16:33:25
	• Maintain LOS	
	• Attitude db = $\pm 10^{o}$/3 axes	
	TPI MANEUVER	05:18:17:11
	• Posigrade, Pitched Up	
	• Attitude db = $\pm .5^{o}$/3 axis	
36	• Audio	
pp 4-29	TPI BURN	05:18:28:11
	• OMS	
166	• ΔV = 21 f.p.s.	
pp 2-9	• Apogee/Perigee = 171/160 N.M.	
	• Target Range < 300 n.m.	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	TPF MANEUVERS	05:19:00:56
	• RCS	
	• Attitude db = ±.5^{o}/3 axis	
	• Maintain LOS on Target	
	• Apogee/Perigee = 170/170 N.M.	
	• $\Delta \dot{V}$ = 28 f.p.s.	
	• Audio/Visual Cues	
	FINAL CLOSURE	05:19:20:56
	• Translate to 500 feet at 1 f.p.s.	
	• Activate Manipulator Arms	
36 pp 4-29	• Orient Below Target to 50 ft. at 1 f.p.s.	
	• Retrieve Payload with Manipulators	
	• Secure Payload and Manipulators	
	• Audio/Visual Cues	
	BURN ATTITUDE AND HOLD	06:19:57:23
	• Attitude	
	• Powered Flight Navigation	
	• IRU Alignment	
	• db = ±.5^{o}/3 axis	
36 pp 4-30	START APU	
	SECURE RADIATORS	
	SECURE P/L BAY DOOR	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	SYSTEMS CHECKS	
	DEORBIT BURN	06:20:17:23
36 pp 4-30	• OMS	06:18:31:26
	• ΔV = 250 f.p.s.	
	• Apogee/Perigee = 170/TBD	
	• Audio/Visual Cues	
	ENTRY PREPARATIONS	
	• Entry Attitude-Orbital Rate	
	• IRU Alignment	
	• System Status Checks	
	• OMS Propellant Dump-Non Propulisve	
	• Targeting	
	ENTRY INTERFACE	06:20:37:43
46	• Altitude ≅ 400,000 feet	
	• Angle of Attack = $+34^{0}$	
36	• Roll Angle = 85^{0}	
	• V = 25,700 f.p.s.	
	• GAMMA = -1.0^{0}	
166 pp 2-9	• Monitor Heat Loads, Trajectory	
	• Audio/Visual/Motion Cues	
	• Weight = 195,583 lb.	
	VHF COMM BLACK-OUT STARTS	06:20:40:43

KEY REF	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	VHF COMM BLACK-OUT ENDS	06:21:08:43
4	o Check Navigation Errors	
pp 9.12c-2	o Altitude = 130,000 feet	
	LOCK ON RADIO NAVIGATION	
	o TACAN/ILS/Radar Altimeter	
	HIGH KEY	06:21:09:43
	o Begin Automatic Terminal Guidance	
	o Initiate Transition Maneuver	
	- Begin Angle of Attach Transition to Front Side of L/D Curve	
	o Energy Management	
	o End Transition Maneuver	06:21:12:43
	o Pre-Landing Checklist	
	ABES IF APPLICABLE	
	o Checklist	
	o Engine Start	
	o Altitude = 40,000 feet, Mach = 1.0	
	LOW KEY	06:21:18:23
	o Altitude = 12,000 feet	

KEY REF.	EVENT/ACTIVITY KSC LAUNCH-EASTERLY	GET d h m s
	o Turn to Final Approach Heading	
	o Energy Management	
	FINAL APPROACH	06:21:18:43
	o Glide Slope	
	o Modulate Speed Brakes and Angle of Attack	
	o Localizer	
	LANDING FLARE	06:21:20:33
	o ILS Glide Slope	
	o Altitude = 500 feet	
	o Landing Gear Down	
	TOUCHDOWN	
	o KSC Landing	
	o Speed = 152 Knots	
	o Deploy Drag Chute, Speed Brakes	
	o Wheel Brakes and Steering	
	o Latitude = +28.567^{0}	
4 pp 9.12c-12	Longitude = -80.617^{0}	
	Altitude = 15 feet	
	Heading = 150^{0} or 330^{0}	
166 pp 2-9	o Runway 300' X 15,000'	
	o Weight = 193,168 lb.	

2.1.6.2 Data References

8

10

21

26

36

46

57

166

KEY REF.

2.1.7 Aborts

1 The Shuttle Orbiter will possess intact (crew, payload, vehicle) abort capability throughout the boost phase. Return will be to the initial launch site. Crew-only safe egress will also be provided on the pad. Candidate abort regions which have been identified are:

Pad and low altitude

Return to site (unpowered)

2 Return to site (powered)

Once-around orbit

To orbit-degraded mission

Booster powered glide return

Data References

1. 20 P. IV-6
2. 7 P. A.7-1; 16 p. 2-16; 25 p. 42

2.1.7.1 Pad and Low Altitude

2.1.7.1.1 . Requirements

1 This abort region extends from the pad to the point in the boost trajectory at which an unpowered glide return becomes possible (about 30 seconds).

Two abort procedures are foreseen in this region:

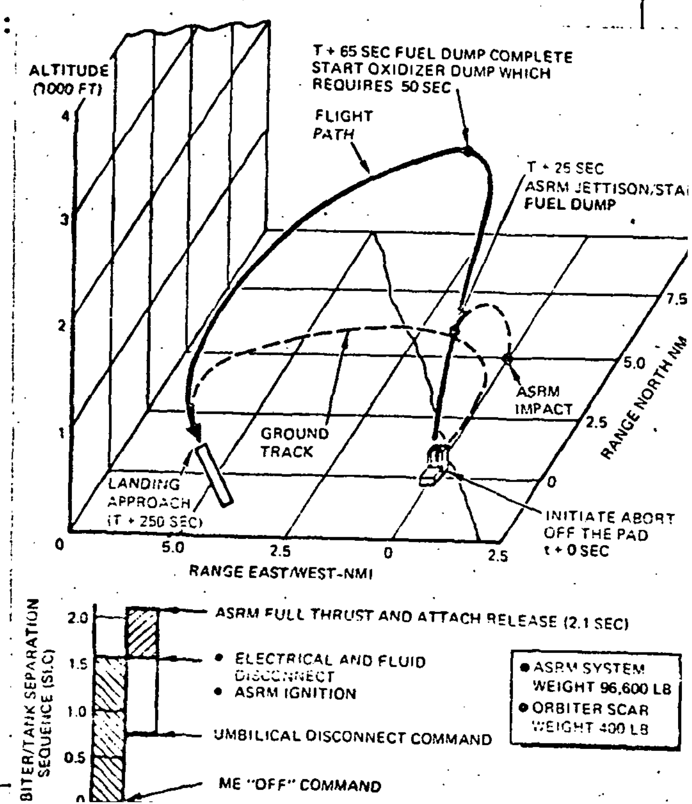
2 1) Crew Egress. If launch commit has not yet occurred and warning time is adequate (e.g., 2 minutes), the crew will leave the orbiter, and proceed to blast shelters via high speed elevator or slide wires to a blast-protected room.

3 2) Orbiter abort. If launch commit has occurred or warning time is short, two special Abort Solid Rocket Motors (ASRM) will be used.

KEY REF.

After ASRM ignition, booster thrust is terminated, the orbiter separated
4 from booster and external tank, and the orbiter rides the abort rocket to burnout. *(An alternate strategy suggested would be to remain attached to the tank, allowing use of the orbiter main engines as well. The tank would be jettisoned at a safe altitude after burnout.)*

Aerodynamic control is expected to be adequate, and no abort SRM TVC is required. Accelerations should not exceed 3 g's ASRM's must be jettisoned upon burnout. OMS fuel and oxidizer must also be dumped after ASRM burnout to facilitate orbiter trim for landing. The orbiter will execute an unpowered glide return to the launch site during and after propellant dump. The ASRM(s) will provide an initial thrust-to-weight ratio of about 2.45 (full payload), burn time of about 21 seconds, and a total impulse of near 16 million lb-sec. It is anticipated that the ASRM's will be jettisoned unburned on a nominal mission shortly after this abort
5 region ends. Trajectory information for ASRM aborts at 0 seconds (pad) has been developed and is shown below:



KEY
REF.

2.1.7.1.2 Rationale for Assumptions

Not applicable

2.1.7.1.3 Data References

1) 166 pp 2-49, 2-50

2) 7 p. A. 7-2; 25 p. 39; 20 p. IV-6; 13 p. 4-7; 162 p. 2-49

3) 7 p. A. 7-2, A. 7-3; 16 sect. 6; 166 p. 2-50

4) 13 p. 4-7, 4-8, 4-9

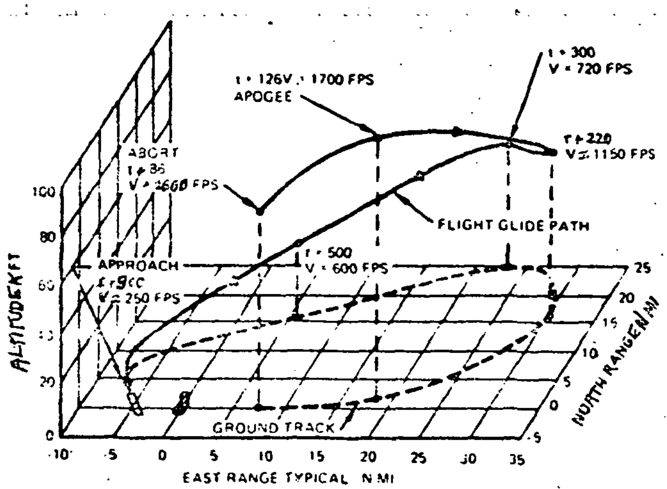
5) 166 p. 2-50, 3-81, 3-82

REF KEY

2.1.7.2 Return to Site Unpowered

2.1.7.2.1 Requirements

1 The glide return abort region begins at the point at which the vehicle has attained altitude and velocity permitting unpowered return. It is expected to last from 30 seconds after launch to 86 seconds after
2 launch, after which downrange momentum is too great. Upon abort in this region, thrust is terminated, and the orbiter separated from the external tank/SRM cluster. It is expected that separation can be accomplished
3 aerodynamically in this region; (maximum dynamic pressure is attained in this interval). After separation, the orbiter executes an unpowered glide back to the launch site. A trajectory for such an abort at 86 seconds
4 after launch has been developed, it involves a coast to apogee, followed by a turn to return to the launch site. It is illustrated below:



2.1.7.2.2 Rationale for Assumptions

Not applicable

KEY
REF.

2.1.7.2.3 Data References

1) 7 p. A. 7-2; 16 p. 2-18; 25 p.42; 166 p. 2-51

2) 7 p. A. 7-3, A. 7-4; 16 p. 2-17; 166 p. 2-51

3) 166 p. 2-51

4) 7 p. A. 7-5; 16 p. 2-19; 166 p. 2-51

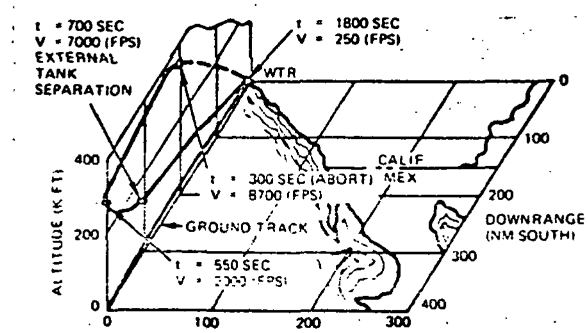
REF KEY

2.1.7.3 Return to Site Powered

2.1.7.3.1 Requirements

1 This abort region begins at the point at which range and velocity do not permit an unpowered glide return, nominally about 86 seconds after launch. Staging is ordinarily included in this region. The region terminates at the point at which all the HØ in the external tank is required to create an acceptable return trajectory. If an orbiter engine out is the malfunction prompting abort, this occurs about 300 seconds after launch. Upon abort, the booster SRM thrust is terminated and the SRM's are separated (if attached). The orbiter main engines continue thrusting, and a retrograde attitude is assumed to brake downrange velocity, then propel the orbiter back toward the launch site. After reaching a point at which an unpowered glide return can be accomplished, the external tank is released (at a dynamic pressure between 2 psf and 25 psf) and a normal approach and landing is made at the launch site. A trajectory has

2 been developed for abort at the end of the region (300 sec. after launch). Trajectory information is:



2.1.7.3.2 Rationale for Assumptions

Not applicable

2.1.7.3.3 Data References

1. 7 pp. A. 7-2, A. 7-5, A.7-6; 16 pp. 2-17, 2-18, 2-19; 25 p. 42; 166 p. 2-51
2. 7 pp. A.7-6, A.7-7; 16 p.2-20; 166 pp. 2-51, 2-52

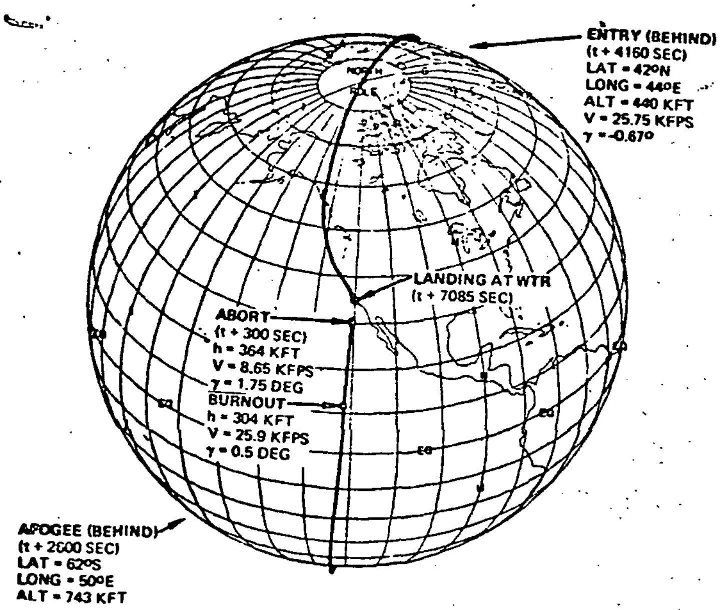
REF KEY

2.1.7.4 Once-Around Orbit

2.1.7.4.1 Requirements

1 This region begins at the point at which the orbiter can inject into a once-around orbital trajectory, returning to the launch site without a deorbit burn (e.g., a 16 x 176 n.mi. orbit). The Orbital Maneuvering System and RCS is used to supplement the orbiter main engines in providing thrust to attain once-around orbital conditions. By using

2 109% EPL on the main engines, the once-around region's beginning coincides with the end of the powered glide return region. It is expected to end about 140 seconds later when insertion into a nominal orbit is feasible. The insertion conditions must be targetted so as to ensure acceptable entry V-γ conditions making provision for navigational uncertainties, and so as to maintain required downrange from the orbiter's entry interface within the orbiter's aerodynamic capability. The once-around region is highly dependent on trajectory conditions, OMS propellant available, and nature of the malfunction causing the abort. The following trajectory parameters have been developed for a once-around abort at 300 sec. into a polar orbit boost:



2.1.7.4.2 Rationale for Assumptions

Not applicable

2.1.7.4.3 Data References

1. 7 pp. A. 7-6, A. 7-7; 16 pp. 2-20, 2-21; 166 pp. 2-51,2-52
2. 7 p. A. 7-2, 16 p. 2-18
3. 166 p. 2-52

Ref.
Key

2.1.7.5 To Orbit-Degraded Mission

2.1.7.5.1 Requirements

1 This region begins at the point at which the orbiter can inject into a 50x100 n.mi. orbit with adequate OMS propellant to circularize and deorbit (about 350 ft/sec). The remaining OMS propellant above this amount may be used in the boost burn. Some of the mission objectives may be accomplished in this case.

2.1.7.5.2 Rationale for Assumptions

Not applicable

2.1.7.5.3 Data References

1) 7 p. A.7-7, 16 p. 2-21; 25 p. 4-2; 166 p. 2-52

Ref. Key

2.1.7.6 Booster Powered Glide Return

2.1.7.6.1 Requirements

1 A booster powered glide return is an optional abort mode available if an orbiter main engine fails during mated ascent. If other abort modes are not preferred, the mated burn can be continued until ordinary shutdown of the booster, whereupon an ordinary glide return abort can be executed.

2.1.7.6.2 Rationale for Assumptions

Not applicable

2.1.7.6.3 Data References

1) 7 p. A. 7-8, 166 p. 2-51

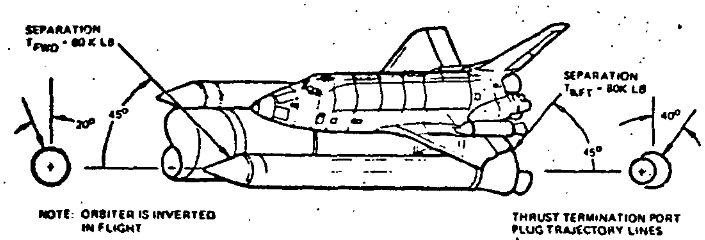
REF.
KEY

2.1.8 Mission Operations

2.1.8.1 Staging

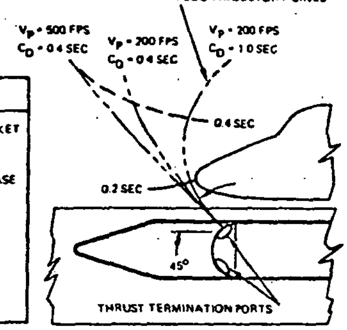
2.1.8.1.1 Requirements

1 Staging, expected to occur 109.1 seconds following launch,
2 is initiated after SRM thrust falls below 5% of nominal average value. Separation is accomplished by use of special separation rockets emplaced in the SRM casings. Three rockets are placed forward, three aft. Each rocket has an average vacuum thrust of 27,000 lb. Aft separation rockets are installed at a greater angle than forward rockets to compensate for the 17° SRM engine cant angle. 0.02 ± 0.01 seconds following the firing of separation rockets, pyros fire to release SRM attachments to the external tank. The separation rockets continue to burn for 2 seconds. During abort staging, firing of separation rockets is delayed until 3 seconds following SRM thrust termination. The sequence is illustrated below:



SEPARATION SEQUENCE

ABORT	NOMINAL STAGING
t = 0 SEC THRUST TERMINATION (T_{SRM} ~ 10%)	t = 0 SEC INITIATE ROCKET FIRING AT T_{SRM} < 5%
t = 3 SEC FIRE ROCKETS	t = 0.02 SEC ±0.01 RELEASE SRM
t = 3.02 SEC ±0.01 RELEASE	
t ~ 5 SEC SEPARATION ROCKET BURNOUT	t ~ 2 SEC SEPARATION ROCKET BURNOUT
t ~ 7 SEC SEPARATION COMPLETE	t ~ 4 SEC SEPARATION COMPLETE



2.1.8.1.2 Rationale for Assumptions

Not applicable

2.1.8.1.2 References

1. 166 p. 2-17; 7 p.A.6-6
2. 166 pp.2-46, 2-47, 3-80, 7-81
   7 pp. A.6-26, A.6-27, A.6-28, C.1-19
   16 pp. 5-16, 5-17, 5-18, 5-19
   25 p. 34

REF KEY

2.1.8.2 Tank Drop Maneuver

2.1.8.2.1 Requirements

The external tank will be jettisoned after insertion. Elapsed
1 time from insertion to tank jettison ranges from 5 minutes to over 20
minutes, depending upon the mission. The process begins shortly after
2 insertion, when dumping of remaining propellant is initiated. O_2 will be
dumped first, followed by H_2. Tanks will be depressurized to 5 to 15 PSI
before jettison. The dumping process will be manually controlled.
3 Propellant disconnects are then separated, to prevent disturbing force
at tank separation. Next, the orbiter is oriented to the tank deorbit
1 4 attitude (ordinarily about 40° above velocity vector - varies from 37°
for deorbit at 100 n.mi. alt. to 67° at 50 n.mi. alt). Attitude is
3 maintained within $\pm 1^{\circ}$. The tank is separated with a zero-force release,
and the orbiter flies away with an 8 second RCS burn which imparts a
$3 \frac{ft}{sec}$ relative velocity along the orbiter z-body axis. About 17 seconds
thereafter, the deorbit motor on the external tank ignites, and burns for
37 seconds with a thrust of 18,500 lb. The result is a tank ΔV of
$300 \frac{ft}{sec}$, adequate to deorbit it.

2.1.8.2.2 Rational for Assumptions

Not applicable

2.1.8.2.3 References

1. 166 p. 29
2. 166 pp. 3-60, 3-61
3. 166 pp. 2-45, 2-46
4. 166 p. 2-10

REF KEY

2.1.8.3 IRU Alignment

2.1.8.3.1 Requirements

1 IRU alignment will be accomplished periodically during the flight and, in addition, shortly before certain critical events. Ordinarily
2 the on-board computer will schedule IRU alignment, although the crew may, in unusual cases, also request IRU alignment. Fine alignment will be conducted entirely by the computer, ordindarily without manual assistance, up to the point of actual gyro torqueing. The computer will drive the star trackers, and issue its own "mark" commands. It will perform reasonableness tests on the results, and check computed IRU errors against expected errors. If results are unreasonable, the crew will be notified. If results are reasonable, the computer will display calculated gyro torqueing angles and expected gyro torqueing angles for acceptance. A
3 CØAS will be provided for IRU coarse align (and backup rendezvous angle tracking).

2.1.8.3.2 Rationale for Assumptions

Not applicable

2.1.8.3.3 References

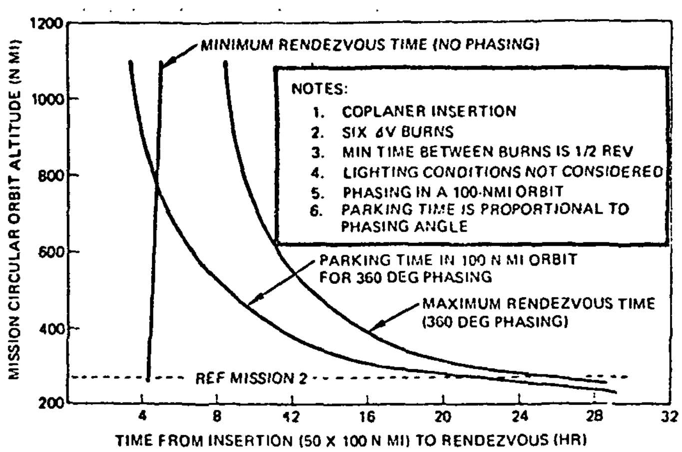
1. 46 3-14 through 3-18; 30 pp. 9.6-96, 9.6-97
2. 30 pp. 9.6-94 through 9.6 - 109
3. 21 p. 29; 166 p. 3-97

REF KEY

2.1.8.4 Rendezvous

2.1.8.4.1 Requirements

Unlike Apollo or Skylab, a shuttle rendezvous sequence can
1 contain any number of burns. The on-board computer will plan the rendezvous sequence, and target the individual burns, for rendezvous with a coopera-
2 tive target vehicle. The use of ground tracking facilities is anticipated for rendezvous with a passive satellite. The precise sequence used in a
1 given case will be dependent on crew specified constraints and the initial
3 relative states of the two vehicles. A coelliptic burn sequence will
1 ordinarily be used. The orbiter will ordinarily be the active vehicle during rendezvous. The ØMS will normally be used to provide delta-V
4 through terminal Phase initiation, after which the RCS will provide mid-course and braking thrust. For single orbit retrieval missions, it may
3 be necessary to burn the TPI burn during boost. (i.e., at insertion, the orbiter would be 10 n.mi. below and 25 n.mi. slant range behind the target). In such case, the ØMS would be used for major braking in the target vicinity (about 110 $\frac{ft}{sec}$ delta-V). Braking burns will be targetted and effected by
5 the on-board computer. Rendezvous flight times have been estimated for
6 a number of orbital altitudes, as shown below:



3 Apollo A-steer is used for guidance steering. Navigation updates during the latter stage of rendezvous with a cooperative vehicle are obtained using TACAN equipment, and a star sensor. TACAN provides range data and the star sensor acquires a target light to provide bearing. Range rate (to within .5 $\frac{ft}{sec}$) will be obtained by the on-board computer from range data.

2.1.8.4.2 Rationale for Assumptions

Not applicable

2.1.8.4.3 References

1. 20 p. IV-9; 4 pp. 9.8-1 through 9.8-16
2. 166 pp. 3-97, 3-110
3. 166 p. 2-79
4. 20 pp. IV-S, IV-19, IV-20
5. 30 pp. 9.8-S6 through 9.8-69
6. 166 p.2-75

REF KEY

2.1.8.5 Docking

2.1.8.5.1 Requirements

1 Docking will be manually controlled. RCS will be used for
2 attitude maintenance and translational propulsion. The payload manipulators may also be used in the final phase of docking, for relative state
2 3 sensing, maneuvering, and braking. The docking mechanism, located behind the cockpit on the upper side of the vehicle, provides for positive engagement and release of the target, and need not be removed for personnel and cargo transfer. Maximum docking misalignments and errors are:

3

lateral misalignment	±.5 ft
angular misalignment	$\pm 5^{\circ}$
roll misalignment	$\pm 7^{\circ}$
closing velocity	$.5\ \frac{ft}{sec}$
active vehicle rates	$1\ \frac{deg}{sec}$
passive vehicle rates	$.1\ \frac{deg}{sec}$

4 Time from rendezvous completion (within 10 n.mi.) to docking may be as short as 5 minutes.

2.1.8.5.2 Rationale for Assumptions

Not applicable

2.1.8.5.3 References

1. 166 p.3-96
2. 30 pp. 9.10-1, 9.10-2; 20 p. IV-18, 22 pp II-1, VIII-9, VIII-6
3. 166 pp.3-52, 3-53
4. 166 p. 2-9

REF KEY

2.1.8.6 Payload Deployment

2.1.8.6.1 Requirements

1 Payload deployment is accomplished using two 50-foot arm-like cargo manipulators. Only one of the two arms is required for ordinary cargo handling operations. (Both arms are needed for orbital assembly.) After opening the payload bay doors, the arms are deployed above the vehicle fuselage. Each arm possesses a shoulder, elbow, and wrist, with 2, 2, and 3 joints, respectively. A torque-producing electric motor is located at each joint. Arms will be manually controlled using hand or arm controllers in the crew station with variable gain force feed back. Manual commands will be translated to torque commands for joints in the on-board computer. Visual information will be provided to the operator by direct viewing from the cockpit, supplemented by four TV cameras. One camera is mounted on the end of each arm, one forward, and one aft
2 in the cargo bay. The fifth is aligned along the centerline of the
3 docking hatch. The payload attachment system is remotely actuated. Payloads are deployed by grasping them with a hand-like terminal device, lifting them out of the payload bay, and releasing them. Residual rates will be less than $.5 \frac{ft}{sec}$ and $.1 \frac{deg}{sec}$. A 65,000 lb. payload can be deployed in 10 minutes. Up to five payloads may be deployed during a mission.

2.1.8.6.2 Rationale for Assumptions

Not applicable.

2.1.8.6.3 References

1. 22 pp. II-2, II-4, VIII-98
2. 7 p. B.3-41; 166 pp. 3-156, 3-159
3. 166 p. 3-156

REF KEY

2.1.8.7 Payload Retrieval

2.1.8.7.1 Requirements

1 Payload retrieval, after successful rendezvous, is accomplished
using the cargo manipulators described in Section 2.1.1.8.7. Once a
2 station-keeping position is attained, a standard manipulator trajectory
for arm deployment to a position short of the payload is commanded and
monitored by the operator. The operator then manually engages the pay-
load, and places it in the payload bay. The manipulator is useful for
1 braking small relative velocities as well as translating the payload. The
3 payload is then secured in the payload bay, and payload interfaces are
connected without EVA.

2.1.8.7.2 Rationale for Assumptions

Not Applicable

2.1.8.7.3 References

1. 22 pp. II-2, II-4, VII-98
2. 166 p. 3-159
3. 166 p. 3-156

REF KEY

2.1.8.8 Deorbit

2.1.8.8.1 Requirements

The deorbit burn will place the orbiter on a trajectory which will result in a safe (tolerable heating and g-loads) entry and
1 landing at the desired place. The deorbit burn will be targeted by the on-board computer. Since, in general, several acceptable deorbit opportunities will exist, the on-board computer will display alternative deorbits to the crew, which will select a particular opportunity based on entry crossrange, time to ignition, required delta-V, lighting conditions at landing, urgency of return, etc.

The following table provides a summary of deorbit opportunities for the reference missions.

2

Launch Site	Orbit Altitude Inclination	Ascending Node Opportunity	Wait* Time (Hrs)	Descending Node Opportunity	Wait** Time (Hrs)
ETR	100 nm/28.5°	4	0***	3	15
ETR	270 nm/55°	3	6	3	12
WTR	100 nm/90°	2	10.5	2	10.5

*Time between last ascending node and first descending node opportunity
**Time between last descending node and first ascending node opportunity
***Seven consecutive opportunities

3 A single ØMS burn will provide the required deorbit delta-V. Sample
4 values of deorbit Delta-V range from 230 $\frac{ft}{sec}$ in a 100 n.mi. circular
5 orbit to 800 $\frac{ft}{sec}$ in a 500 n.mi. circular orbit. Deorbit guidance will be provided by an adaptive guidance system, necessary since burn arcs of up to 70° will be possible.

2.1.8.8.2 Rationale for Assumptions

Not applicable

2.1.8.8.3 References

1. 31 p. 9.12-4
2. 166 p.2-75
3. 20 p. IV-20; 166 p.1-3
4. 166 p.2-8
5. 166 pp. 2-79, 2-80

REF. KEY

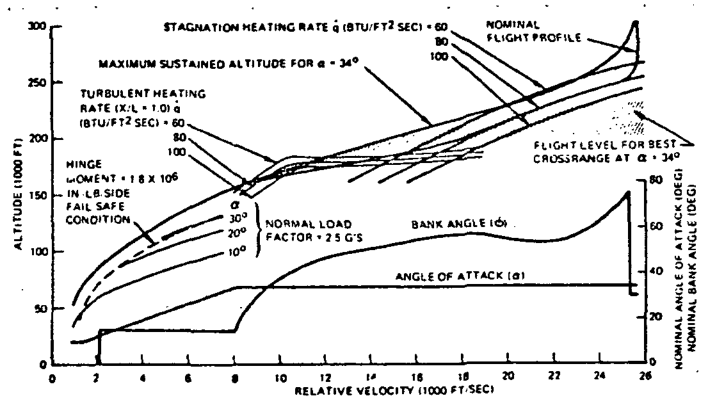
2.1.8.9 Entry

2.1.8.9.1 Requirements

1 Following the deorbit burn, RCS is used to orient the orbiter to the entry attitude. Entry attitude for the high cross-range polar trajectory has an angle of attack of 34° and a bank angle of 30°. An adaptive initial entry guidance law is used to compensate for deorbit, aerodynamic, and atmospheric dispersions. Angle of attack is held constant and bank angle is modulated to control heating rate until a pullout
2 stagnation heating rate has been attained (80 $\frac{BTU}{ft^2\ sec}$ normally). The
3 pullout heating rate is then maintained by modulating bank angle until an altitude of about 215,000 feet is achieved. The next sector of the entry profile is flown along a descending path at a constant rate of change of flight path angle of -.00025 deg/sec until nominal bank angle reaches 15°, after which time it is maintained until pitchdown.
1 This is approximately the minimum altitude trajectory that satisfies turbulent heating constraint. Bank angle is modulated to control heating and cross-range. Angle of attack is modulated for downrange control.
3 4 The automatic guidance system will use a stored reference trajectory, and add angle of attack and bank angle perturbations to the trajectory schedule for off-nominal conditions. Scheduled angle of attack for the
1 entry (until pitchdown) is a constant 34°. Variations from the schedule of up to 4° in angle of attack are anticipated as the guidance system corrects for range-to-go feedback. Entry constraints, and normal trajectory and schedules are shown below:

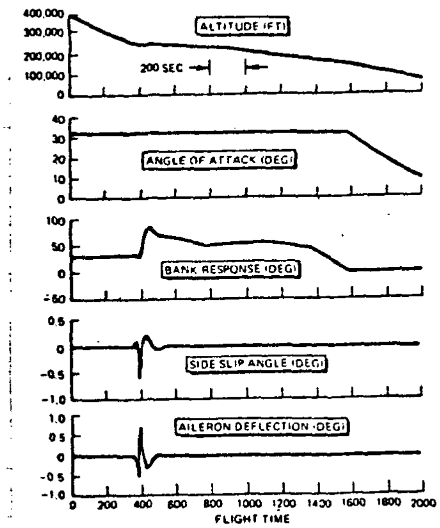
REF.
KEY

2



5 Three-axis RCS attitude control will be used exclusively until dynamic pressure sensed by vehicle acceleration exceeds 2 $\frac{lb}{ft^2}$. At that time, trim commands to the elevons are initiated. Attitude control and rate damping functions are still performed by RCS. When dynamic pressure exceeds 25 $\frac{lb}{ft^2}$, elevons assume pitch and roll control, replacing RCS. In the automatic mode roll control for bank angle modulation is obtained by differential deflection of the elevons. In manual mode (control stick steering) elevon control is supplemented with 200 millisecond yaw - RCS impulses to improve handling qualities. Elevon trim capability in this flight regime exists for angles of attack between 20° and 50°. In manual modes, the pilot corrects bank angle and angle of attack steering errors. Yaw attitude control is performed by RCS. Entry control profiles are shown below for the automatic system.

REF.
KEY



1 On the high cross-range polar trajectory, about 23 1/2 minutes elapse between entry interface (400,000 ft. altitude) and the beginning of pitchdown (about mach 8).

2.1.8.9.2 Rationale for Assumptions

Not applicable.

2.1.8.9.3 References

1. 166 PP. 2-77, 2-80
2. 166 P. 2-76
3. 167 PP. 1-4, 1-5
4. 4 PP. 9.12D-1 through 9.12D-11; 31
   PP. 9.12-48 through 9.12-55
5. 166 PP. 1-4, 2-61, 2-70, 2-71

2.1.8.10 Hypersonic-Supersonic Transition

2.1.8.10.1 Requirements

1 When velocity reaches 8,000 ft/sec, a gradual pitchdown maneuver is initiated, from an angle of attack of 34° to an angle of

REF. KEY

attack of about 10°. The pitchdown maneuver will ordinarily last 7-8 minutes, and will terminate at a velocity of about 1,500 ft/sec. The angle of attack rate will be approximately constant, while bank angle will be modulated for downrange and cross-range control. Bank angle

2 will be restricted to 60°. As in the case of entry, guidance and control will be derived from the on-board computer, with manual backup.

3 Control will be effected as for entry (pitch and roll controlled by elevons, yaw by RCS) until angle of attack drops below 20° and mach number below 4. At that point, the rudder will be activated, and yaw control will be achieved by a blending of rudder and RCS. When angle of attack drops to 10° and mach number is below 2, the rudder is fully effective and yaw-RCS is deactivated.

4 After the communications blackout, TACAN range and bearing information is used to update inertial navigation.

2.1.8.10.2 Rationale for Assumptions

Not applicable.

2.1.8.10.3 References

1. 166 PP. 2-77, 2-80; 167 P. 1-5
2. 26 P. 6-15
3. 166 PP. 2-61, 2-70, 2-71
4. 166 PP. 2-80, 3-97

2.1.8.11 Energy Management

2.1.8.11.1 Requirements

1 After pitchdown is completed, angle of attack is maintained at about 10°, and a wings level trajectory is normally flown until subsonic flight is

REF. KEY

achieved. Angle of attack and bank angle will be modulated to control down-range and cross-range. Approach is ordinarily initiated at an altitude of 50,000 ft. and a velocity of mach 0.8. At this point the split rudder speed brake is opened to 40 degrees. It is thereafter modulated to control indicated airspeed at 250 knots until after final approach initiation. Glide range is controlled by elevons (and the speed
2 brake). The on-board computer will ordinarily fly an optimal two-run approach trajectory, but will fly others if required by initial conditions.
3 Final approach typically begins at an altitude of 10,000 feet with interception of the 15° glide slope. At this time, landing gear is extended. Constant airspeed (250 keas) is maintained until altitude reaches 1,050 feet, where a pullup (preflare) maneuver is initiated to effect a transition to the 3° glide slope. The shallow glide slope is followed until altitude drops to 80 feet, at which time a final flare is initiated to reduce altitude rate to less than 10 ft/sec. Touchdown is targetted to a point 1,500 feet from the runway threshold. The approach trajectory is illustrated below:

REF. KEY

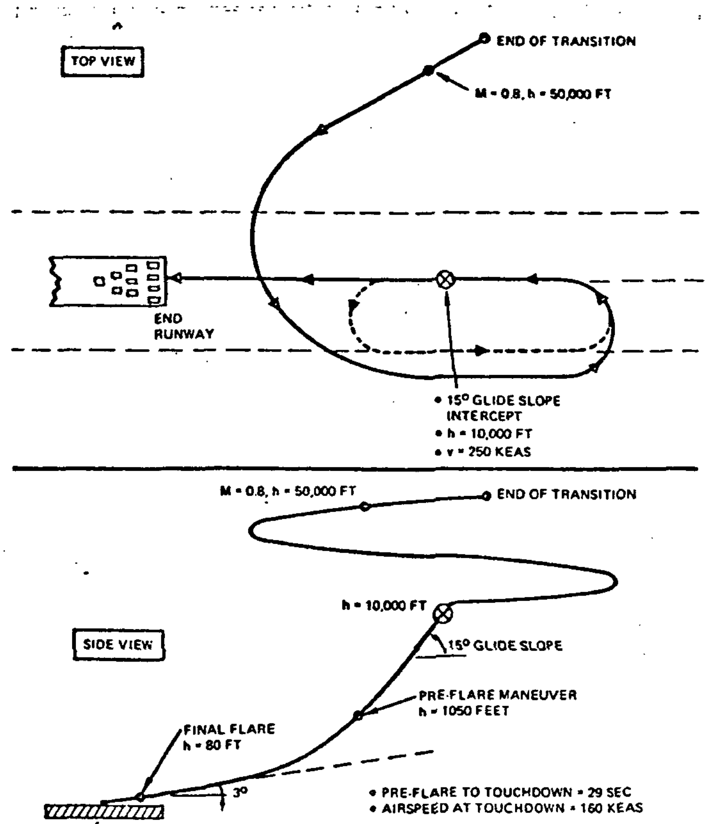
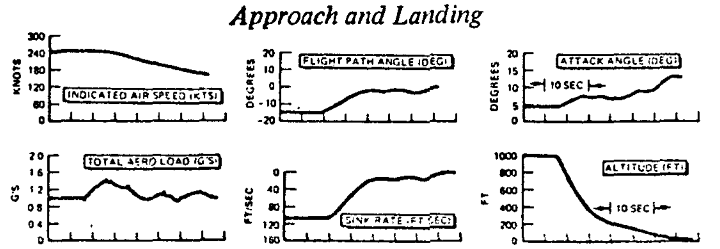


Figure 2-99. Baseline Energy Management Approach and Landing



Control is ordinarily automatic, but manual control is provided. Control stick steering operational modes are expected to be direct, stability augmentation system, and rate command/attitude hold. Glide-brake to elevon trim compensation will offset pitching moment induced by glide brake modulation. Manual inputs will be accepted from rotation controller (two-axis) and rudder pedals. Trim capability exists for angles of attack from -5° to 20°.

REF.
KEY

Continuous navigation updates from TACAN and ILS are accepted by the on-board computer (including two glideslopes). Radar altimeter information is used near touchdown.

1 Twelve minutes is the expected time from the end of pitchdown to landing.

2.1.8.11.2 Rationale for Assumptions

Not applicable.

2.1.8.11.3 References

1. 166 PP. 2-61, 2-77, 2-80
2. 31 P. 9.15-3; 166 P. 3-97
3. 166 PP. 2-72, 2-73, 3-31; 4 P. 9.15-78; 184 P. DP-23
4. 31 P. 9.14-15
5. 166 P. 1-4
6. 166 P. 3-97

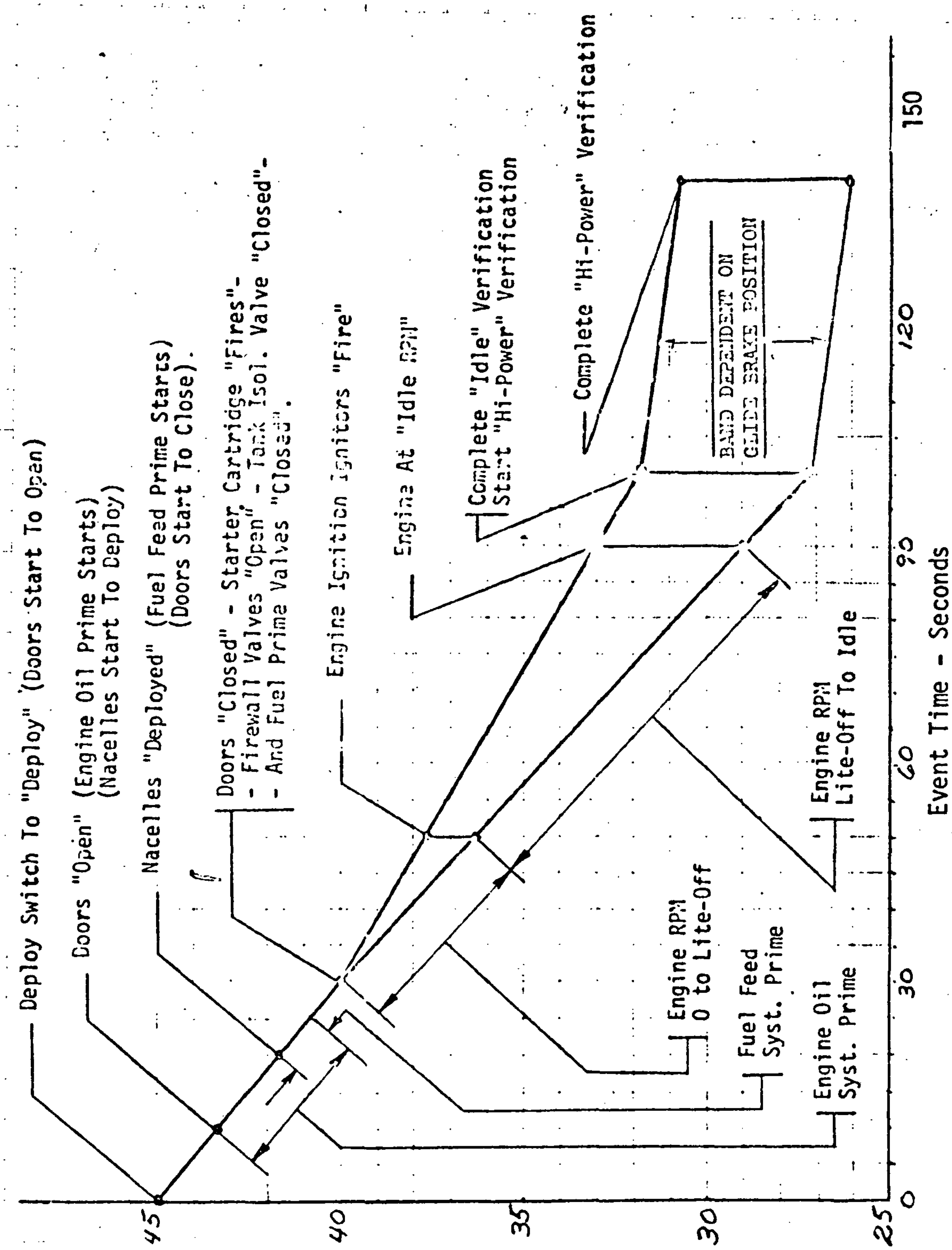
2.1.8.12 Cruise

2.1.8.12.1 Requirements

1 When carried on operational space missions, air breathing engines are deployed below mach 0.9 (by 40,000 feet altitude), started below 40,000 feet, checked out at full power, and returned to idle power stand-by status. Checkout adds 3.5 n.mi. to approach range. The process is illustrated below:

REF.
KEY

2

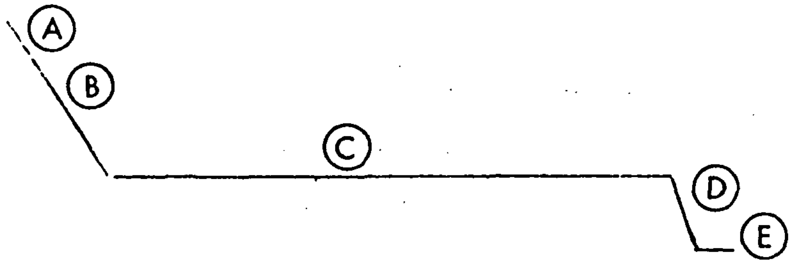

Deploy Switch To "Deploy" (Doors Start To Open)
Doors "Open" (Engine Oil Prime Starts) (Nacelles Start To Deploy)
Nacelles "Deployed" (Fuel Feed Prime Starts) (Doors Start To Close).
Doors "Closed" - Starter Cartridge "Fires"- - Firewall Valves "Open" - Tank Isol. Valve "Closed"- - And Fuel Prime Valves "Closed".
Engine Ignition Ignitors "Fire"
Engine At "Idle RPM"
Complete "Idle" Verification Start "Hi-Power" Verification
Complete "Hi-Power" Verification
BAND DEPENDENT ON GLIDE BRAKE POSITION
Engine RPM Lite-Off To Idle
Engine RPM 0 to Lite-Off
Fuel Feed Syst. Prime
Engine Oil Syst. Prime
0
30
60
90
120
150
Event Time - Seconds
45
40
35
30
25
Altitude - 1000 Ft.

REF. KEY

3 The air breathing engines provide 15 minutes loiter time at 10,000 feet (standard day) to allow operational assessment of conditions prior to
1 landing. Alternately, they may provide a range extension of 97 n.mi. from an altitude of 30,000 feet, or provide one go-around. Two trajectories using air breathing engines are outlined below:

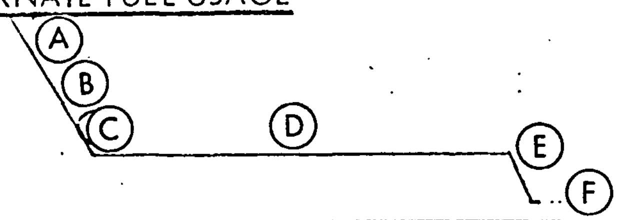
REF. KEY

15 MINUTE LOITER AT 10000 FT



	WEIGHT (LB)	THRUST (%)	FUEL (LB)	TIME (MIN)	RANGE (NM)
(A) ABES DEPLOYMENT AT ~30,000 ft.	193168	-	-	-	-
(B) ABES STARTUP AT 25000 FT & CHECKOUT WHILE DESCENDING TO 10000 FT	193168-192821	0-100	347	5	15
(C) LOITER - MAX ENDURANCE AT 10000 FT	192821-179521	78	13300	15	56
4 (D) DESCENT - POWER OFF	179521	0	0	2	10
(E) LANDING - POWER OFF	179521	0	0	1	0
	TRAPPED FUEL		34		
	TOTALS		13681	23	81

ALTERNATE FUEL USAGE



	WEIGHT (LB)	THRUST (%)	FUEL (LB)	TIME (MIN)	RANGE (NM)
(A) ABES DEPLOYMENT AT 45000 FT	193168	-	-	-	-
(B) ABES STARTUP AT 40000 FT & CHECKOUT WHILE DESCENDING TO 30000 FT	193168-192821	0-100	347	-	-
(C) INTERMEDIATE POWER DESCENT 30-10K FT	192821-190821	100	2000	9	39
(D) AFTERBURNER CRUISE AT 10000 FT	190821-179721	78	11093	13	48
(E) IDLE POWER DESCENT	179721-179684	IDLE	44	2	10
(F) LANDING & SHUTDOWN - IDLE POWER	79684-179521	IDLE	163	4	0
	TRAPPED FUEL		34		
	TOTALS		13681	28	97

REF. KEY

2.1.8.12.2 Rationale for Assumptions

Not applicable.

2.1.8.12.3 References

1. 167 PP. 1-5, 4-5
2. 167 P. 4-7
3. 166 P. 2-56; 26 P. 7-9
4. 167 P. 4-9

2.1.8.13 Landing and Rollout

2.1.8.13.1 Requirements

1 The shuttle orbiter touchdown angle of attack is normally 13°, yielding speeds (40,000 pound payload) between 142 and 152 knots

2 with ground effects. Sink rate at touchdown will not exceed 10 ft/sec.

The orbiter is decelerated after landing using 70° speed brake deflection, an anti-skid brake system and a drogue parachute. It will be able to stop on a 6,000 foot dry runway, or a 10,000 foot wet runway after clearing a 50 foot obstacle on a hot day (103°F). Rollout

3 control uses the on-board computer and inertial navigation system with continuous ILS localizer updates.

2.1.8.13.2 Rationale for Assumptions

Not applicable.

2.1.8.13.3 References

1. 166 P. 2-52
2. 166 P. 3-31; 167 P. 4-5
3. 166 P. 3-97

F-398-8-A

Ref Key

2.2 Atmospheric Flights

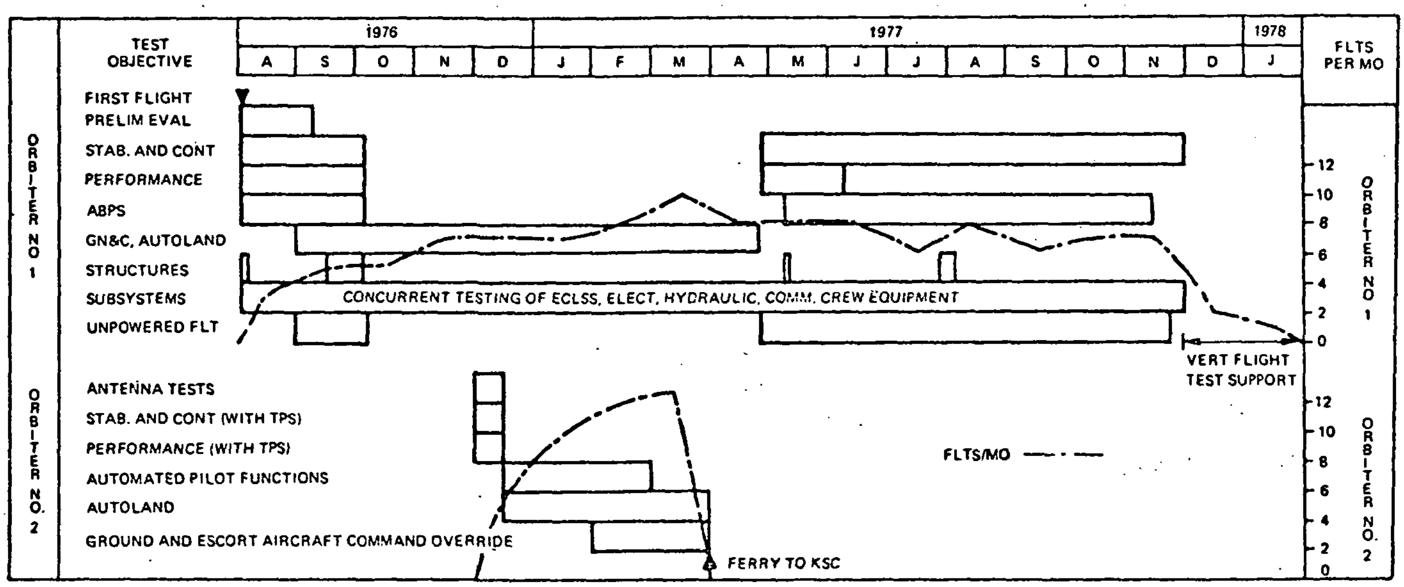
2.2.1 Horizontal Flight Test

2.2.1.1 Requirements

1, 2 The shuttle orbiter flight test program, expected to begin in late 1976, will verify shuttle operation in the horizontal, subsonic, airbreathing portions of the flight envelope. Two orbiters will be utilized in the program. The first orbiter, which will perform most
3 of the testing, will contain only those systems required for horizontal
4 flight and flight testing. The second orbiter will contain all operational systems except **OMS/RCS pods.** Actual configurations are summarized in
2 section 3.3.7.3. The program is expected to consist of 153 flights, and 167 flight hours.

Orbiter 1 will make 113 flights, 137 flight hours, while orbiter 2 will make 40 flights, lasting 30 flight hours. The program will be conducted at Edwards Air Force Base. The program is summarized in the charts below.

4



Ref Key

5

Phase	Horizontal Flight Test Objectives	Dedicated Flight (hr) Orb 1	Orb 2
Preliminary evaluation	Verify basic orbiter flightworthiness:		
	Evaluate low/medium speed handling	7.0	
	Preliminary airspeed calibration		
	Landing gear/low speed airframe buffet		
	Preliminary performance		
	Backup flight controls	3.0	
	Emergency landing gear operation		
	Check systems operation	•	
		10.0	
Flight characteristics • Performance	Determine subsonic performance and demonstrate satisfactory handling qualities:		
	*Takeoff and landing	•	•
	Climb/level speed	8.0	3.0
	Engine-out performance		
	Turning performance		
	Airspeed calibration	2.0	1.0
	Descent/missed approach and go-around	4.0	1.0
		14.0	5.0
• Handling qualities	Static and dynamic stability	23.0	
	PIO susceptibility.		
	Lateral-directional coupling.		
	Speed stability and flight path response.		
	Short period damping.		
	Dutch roll damping.		
	Control response and effectiveness.	22.0	
	Manual trim control.		
	Speed brake response.		
	Crosswind takeoff and landing.		
	Controllability with asymmetric thrust.		
	Stall warning/buffet boundaries.	5.0	
		50.0	
Subsystem performance	Verify subsystem performance in ferry configuration and operational subsonic configuration:		
	ABPS	10.0	
	GN&C	37.0	25.0
	Structures	6.0	
	ECLSS	2.5	
	Crew accommodations	2.0	
	Power	0.5	
	Communications	•	•
		58.0	25.0
Unpowered flight and landing	Verify capability to perform unpowered approach and landing and establish safe initial approach window.		
	Terminal energy management procedures and capability.	1.0	
	Initial and final approach procedure verification and refinement	4.0	
	Approach speed/aim-point optimization.	•	
	Flare control	•	
	Ground effect and float characteristics.	•	
	L/D modulation - speed-brake effectiveness.	•	
		5.0	
	Totals	137.0	30.0

Ref Key

The first orbiter flight will originate at the final assembly site at Palmdale, California, and terminate at Edwards after

6 a brief inflight evaluation of vehicle handling and systems operation. On the initial flight from Edwards, the vehicle's general handling characteristics and subsystem performance will be evaluated, up to 270 knots at 10,000 ft. altitude. The first ten flight test hours will expand the flight envelope to mach 0.7, 25,000 ft. altitude, and normal acceleration to 2 g's. Again, vehicle flying qualities and subsystem performance will be confirmed. Communications and data interfaces with ground stations and mission control will be verified, performance data compiled, and controllability with asymmetric thrust investigated. Remaining horizontal test flights will expand the flight envelope accelerations to 2.5 g's and -1.0 g's. Aerodynamic and response characteristics for various weights, c.g.'s, and configurations (e.g. control surfaces, landing gear) will be established throughout the flight envelope. Takeoff tests will determine optimum nose wheel liftoff speed, high speed ground handling characteristics, crosswind controllability and performance, and engine-out performance. Landing tests will determine runway requirements, go-around performance, optimum braking techniques, and drag parachute performance. Subsystem tests to be performed are summarized below:

SUBSYSTEM TESTS

4

ABPS
- ENGINE CONTROL AND RESPONSE
- INLET DUCT PRESSURE RECOVERY
- INLET DUCT ANTI-ICING
- ENGINE AIR STARTS
- FUEL SYSTEM
- NACELLE AND ENGINE ACCESSORY COOLING
- LUBE SYSTEM

ECLSS
- NORMAL AND EMERGENCY PRESS.
- CREW COMFORT-ELECT/AVIONICS COOLING

COMMUNICATIONS
- VHF AND S-BAND ANTENNA TESTS
- VOICE AND DATA LINKS

ELECTRICAL/HYDRAULIC
- NORMAL AND EMERGENCY POWER
- GENERATION AND DISTRIBUTION

CREW ACCOMMODATIONS
- CONTROLS AND DISPLAYS
- COCKPIT LIGHTING

GUIDANCE, NAVIGATION, AND CONTROL
- AUTOMATIC LANDING
- TACAN, GLIDE SLOPE (15°), LOCALIZER, RADAR ALTIMETER, FLARE AND TRANSITION TO 3° GLIDE SLOPE, TOUCHDOWN, ROLLOUT
- FLIGHT CONTROLS
- RATE SENSING AND BODY BENDING
- STAB. AUGMENTATION SYS REDUNDANCY
- UNMANNED CAPABILITY (MANNED STANDBY)
  - AUTOMATION OF DEDICATED PILOT FUNCTIONS
  - GROUND AND ESCORT AIRCRAFT COMMAND/OVERRIDE

STRUCTURES
- LOADS
- FLUTTER, BUFFETT, DYNAMIC RESPONSE
- LANDING GEAR
- DRAG CHUTE

Ref Key

6 Chase and photo-chase aircraft will be employed throughout all flights which expand the flight envelope or otherwise are judged to require them.

7 MSC Mission Control activity comparable to that of the Apollo program will be conducted for selected horizontal flight tests.

8 NASA Mission Control at Edwards will control most horizontal flight tests.

4 During horizontal flight testing, the orbiters will carry ejection seats and the Development Flight Instrumentation (DFI). The DFI will provide over 2600 measurements of orbiter performance. A

9 dedicated overlay system, it will be removed at the end of the development/testing program.

2.2.1.2 Rationale for Assumptions

Not applicable.

2.2.1.3 References

1. 10 pp. 5-94 through 5-100
2. 178 pp. 7-1
3. 166 pp. 1-12
4. 178 pp. 7-5
5. 178 pp. 7-3
6. 178 pp. 7-4, 7-7
7. 166 pp. 4-1
8. 178 pp. 7-12
9. 178 pp. 7-12; 20 pp. IV-30; 21 pp. 91

Ref Key

2.2.2 Ferry Flights

2.2.2.1 Requirements

1 The shuttle orbiter possesses an operational self-ferry capability. It will be able to fly, via multiple stops or aerial refueling, between the east and west coasts of the continental United States. Propulsion for ferry missions will be provided by four air-breathing engines. A ferry route from Edwards Air Force Base to
2 Kennedy Space Center has been projected. It has stops at:

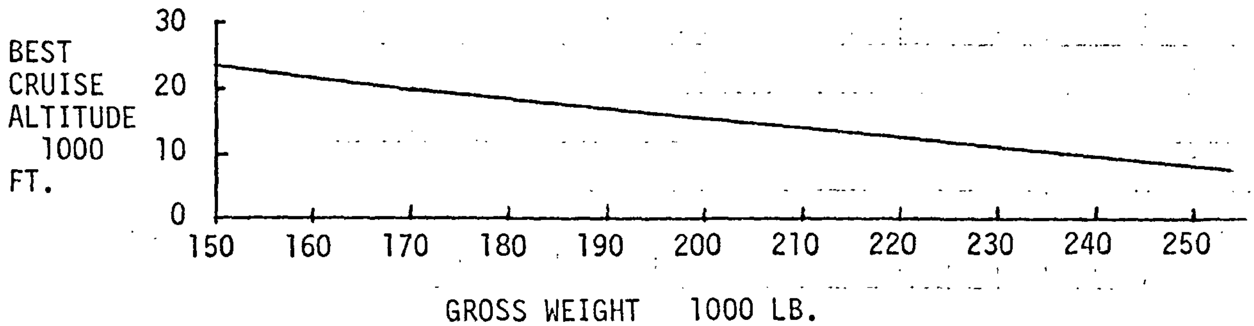
Edwards Air Force Base, California
Davis-Monthan Air Force Base, Arizona
Biggs Air Force Base, Texas
Dyess Air Force Base, Texas
Barksdale Air Force Base, Louisiana
Eglin Air Force Base, Florida
KSC

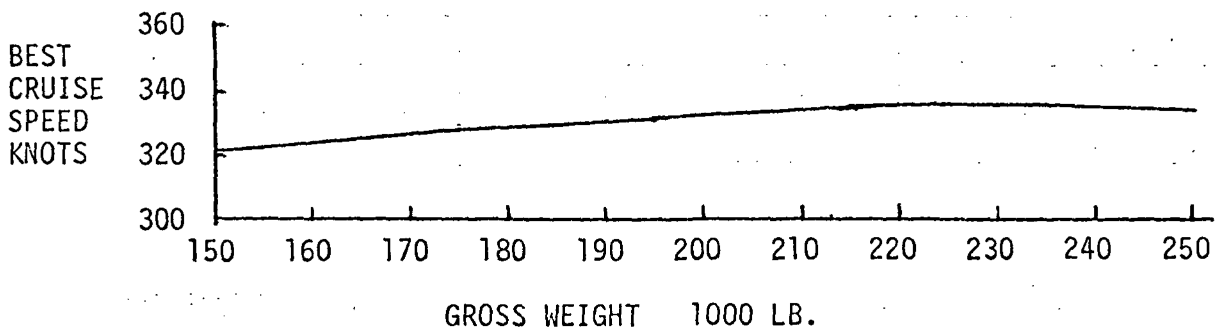
The maximum leg on the route is 386 n. mi., well within
3 the nominal 400 n. mi. ferry range. Elevation of these fields are
4 all less than 4,000 feet.

1 The shuttle orbiter can operate out of a 10,000 ft. long, 150 ft. wide runway (sea level) on a hot (103°F) day. Takeoff weight of
5 216,000 lb. is nominal. Fuel reserve is carried for 20 minutes at idle power plus one minute of maximum power. During takeoff, rotation
6 can be initiated at 168 knots with liftoff at 186 knots. Acceleration to sustained airspeeds greater than 1.2 times stall velocity can be
7 accomplished with gear down. Climb gradient of 5% should be attainable
5 with one engine out on a hot day. Climb is accomplished at maximum

Ref Key 5 continuous power at best climb speed. Cruise is at best cruise speed and altitude with an allowance for a 50kt headwind. Best cruise speed and altitude are shown below.

8 CRUISE CHARACTERISTICS
0007D CONFIG OMS PODS OFF 4F401-PW-400
DRY ENGINES

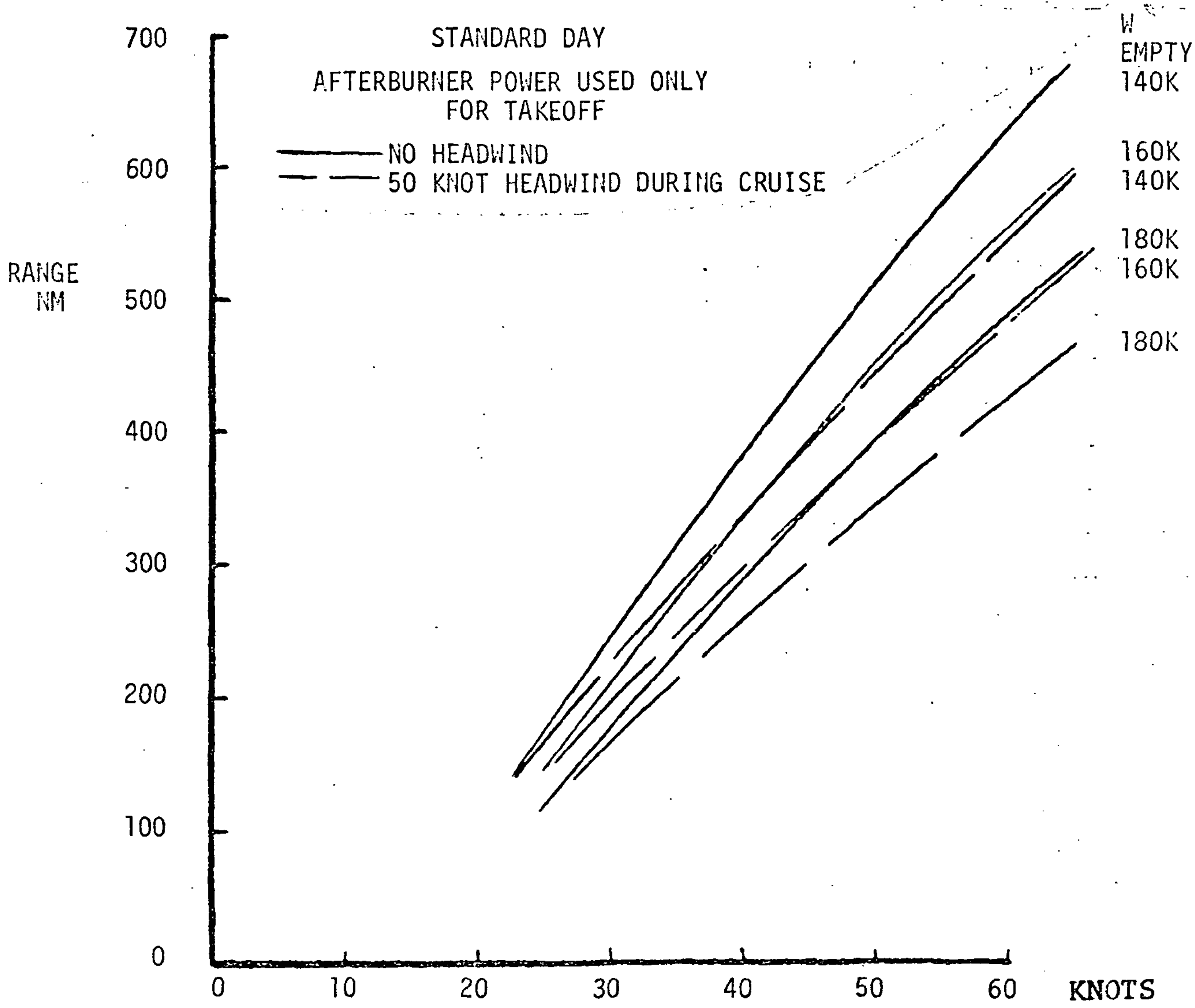




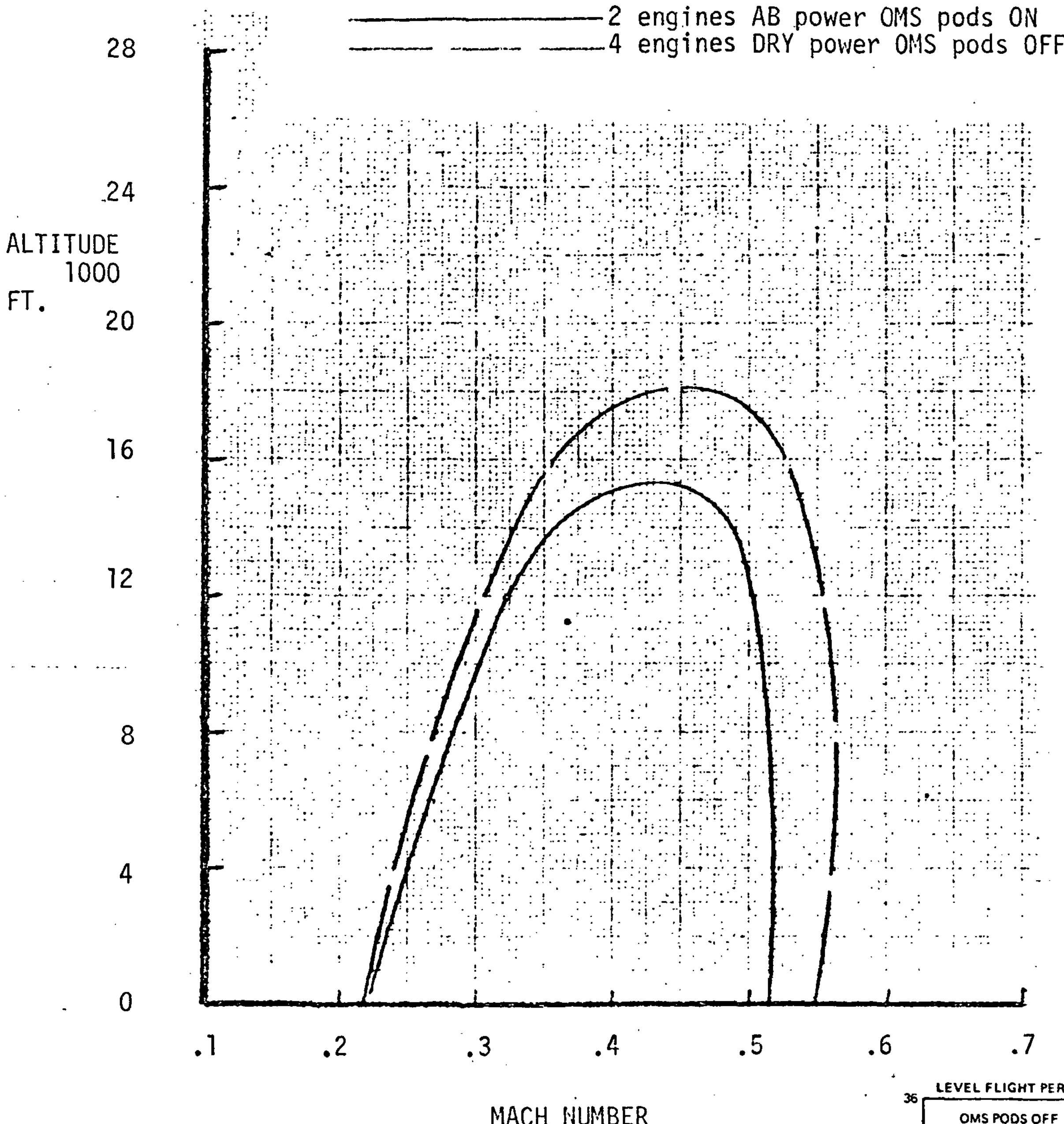
5 Descent is made at maximum L/D in idle power. Fuel reserves are adequate for 20 minutes loiter. The orbiter will be able
1 to land on a 10,000 foot wet runway (sea level) on a hot day (103°F).
9 A summary of a typical ferry mission is shown below.

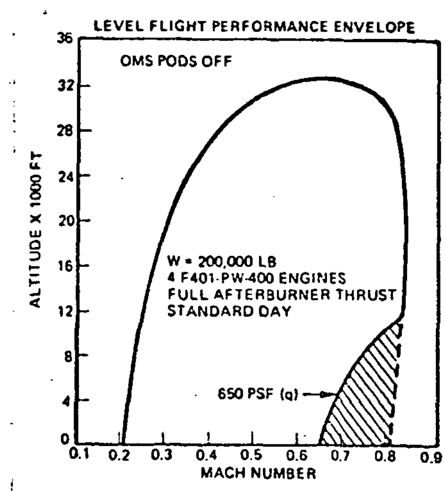
MISSION PHASE	WEIGHT (LB)	ALTITUDE (FEET)	MACH NO.	FUEL (LB)	TIME (MIN)	RANGE (NM)
(A) GROUND	216500-210500	0	0	6000	21	0
(B) CLIMB	210500-202000	0-16000	.39 - .45	8500	15	67
(C) CRUISE	202000-171800	16000-21500	.54	30200	57	311
(D) DESCENT	171800-171500	215000-2000	.42 - .27	300	6	22
(E) LOITER	171500-164700	2000	.32	6800	20	0
(F) APPROACH & LAND	164700-164500	2000 - 0	.3 - .25	200	4	0
			TOTALS	52000	123	400

5, 10 A 50 knot headwind reduces range by about 13%, as shown below.



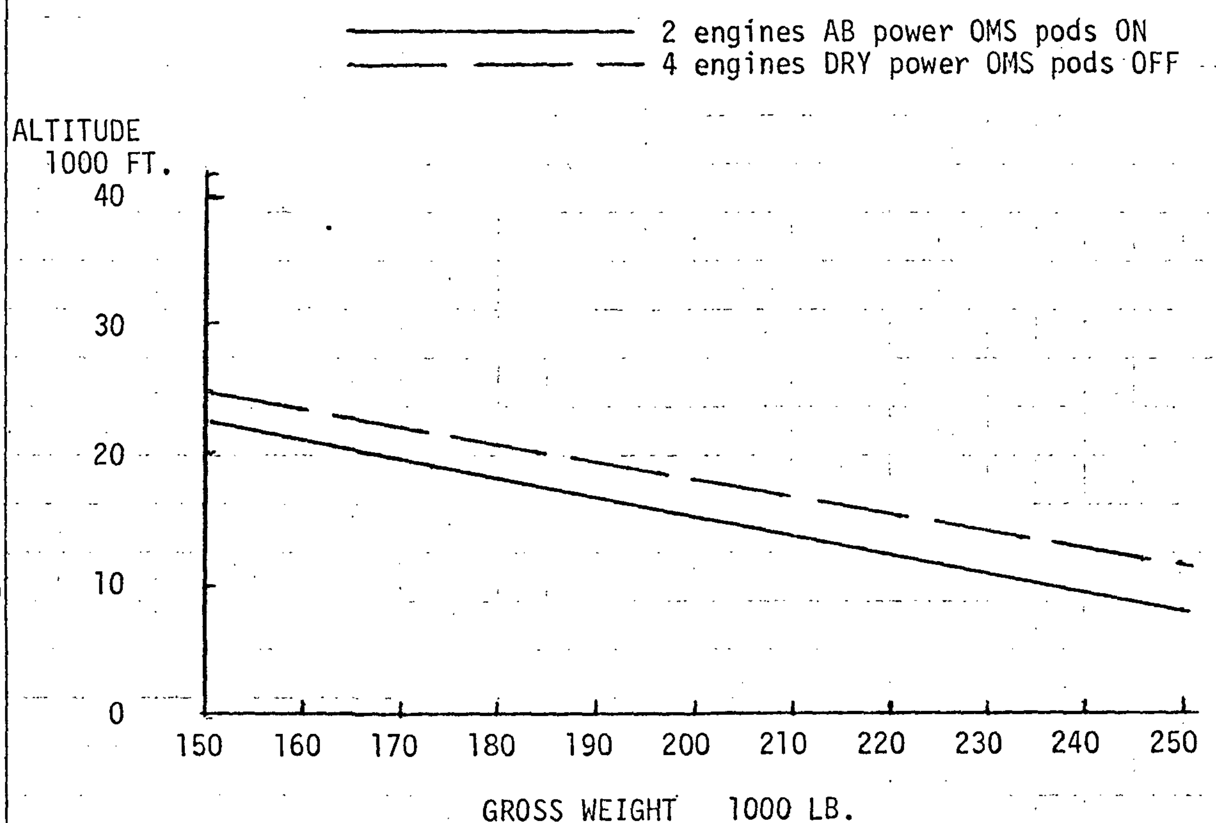
11 Performance envelopes for the airbreathing engines are:





12 Absolute ceilings for various weights are shown below.

ABSOLUTE CEILING



13 The shuttle will be able to cruise at 10,000 feet on a hot day with one engine out.

2.2.2.2 Rationale for Assumptions

Not applicable.

2.2.2.3 References

1. 20 pp. IV-10
2. 166 pp. 4-3
3. 166 pp. 3-77, 3-78
4. 10 pp. 5-102, 5-10F
5. 167 pp. 4-12
6. 166 pp. 2-65

7. 36 pp. 4-252 - see also 166 pp. 2-65

8. 167 pp. 4-15

9. 167 pp. 4-17

10. 167 pp. 4-18

11. 167 pp. 4-21; 178 pp. 7-5

12. 167 pp. 4-23

13. 20 pp. IV-21; 166 pp. 3-79

3.0 Flight Dynamics

3.1 Vehicle Configurations

3.1.1 Operational Space Mission Configuration

3.1.1.1 Orbiter Vehicle

166 The orbiter vehicle described is the NRSSV0007D. The vehicle is delta wing, single vertical stabilizer with three Main Engines (ME), two Orbital Maneuvering System (OMS) engines, and three Reaction Control System (RCS) pods containing 40 engines. The cargo bay is 15 ft. diameter and 60 ft. long located aft of the crew compartment in the upper fuselage. When installed, two Air Breathing Engines (ABE) are installed as a module in the aft end of the cargo bay. External geometry of the orbiter vehicle is **described in Figure 3.1.7.**

GEOMETRY

		Wing	Vertical Tail
	Area	2890	401.8
	Aspect Ratio	2.19	1.675
	Leading Edge Sweep (deg)	50	45
	Diahedral (deg)	+3.5 at T.E.	-
	Incidence (root) (deg)	+3.0 at y = 125	-
167	Incidence (tip) (deg)	-2 at y = 477.4	-
	Span	945.7	311.3
pp 2-4	Chord (root)	720.5	264.7
	Chord (tip)	151.3	107.0
	MAC	497.9	197.0
	X Axis to MAC	186.7	133.6

Elevon Deflection (deg)	+15 -40
MPS Lower Eng. Gimbal	±11° (Null: Pitch = 10.5°, Yaw = 3.5°)
MPS Upper Eng. Gimbal	±11° (Null: Pitch = 16.5 Yaw = 0.0°)
Rudder Deflection (deg)	±15°
SPD Brakes Deflection (deg)	40°
OMS Gimbal	±4° (Null: Pitch = 11.5°, 0 to 12° Yaw = 0°)

	Orbiter Coordinates	X	Y	Z
	Command Pilot Eye	460	-22	469
164	Pilot Eye	460	+22	469
	Docking Hatch	617	0	520
182	Cargo Bay	582 to 1302	0 ± 93	400 ± 93
	MPS Lower Gimbal	1468.3	±52	335.8
	MPS Upper Gimbal	1445	0	433
	OMS	1547	±96	454
	EOHT Attach Aft	1307	±120	-
	EOHT Attach Fore	559.5	0	-
	Deployed ABE	1068 to 1302	±70.5	532.5
	Stowed ABE	1068 to 1302	-	-
	ASRM	1307 to 1550	±168	352

3.1.1.2 Payloads

The payloads configuration is limited to the requirements to fit within the 15 ft. diameter 60 foot length cargo bay. With the ABES installed, the clear cargo bay area is limited to 15 ft. diameter 39 1/4 foot length. Payloads must be configured with the proper attachment hardware for the manipulator arms and with mass and mass distributions to within the

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pp. 2-6

capabilities of the cargo manipulators and mission dynamics constraints. The present conceived payloads will come under general classifications of:

1. Experiment modules
2. Passenger/Cargo modules
3. Cargo modules
4. Propulsive modules (Space Tug.)

All payloads presently conceived are generally cylindrical in shape; however, this is not a known constraint. In addition, if docking is required on the present or future missions, the payload must be equipped with compatible docking hardware. Tables 3.1.1.2.1 through 3.1.1.2.8 list candidate payloads

51
pp. 4-30
4-60

from the Fleming model.

TABLE 3.1.1.2.1

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA ASTRONOMY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	15	LARGE STELLAR TELESCOPE	Extend space astronomy capability to diffraction limited 3 meter dia. optical technology. Deter-mine universe curvature stellar and galactic composition and evolution.	22308	13.1 x 44.9	350/350/28.5
2	17	LARGE SOLAR OBSERVATORY	Conduct high resolution visual and UV studies of solar granular structure and areas of high solar activities. Continue UV and X-ray observations.	27720	15 x 57.1	350/350/30
3	19	LARGE RADIO OBSERVATORY	Understand physical processes in the corona and in the magneto-spheres of the planets, especially Jupiter and Earth.	19998	14.1 x 30.2	350/350/30

TABLE 3.1.1.2.1 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA ASTRONOMY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAYLOAD	IDENT. NOS. FLEMING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
4	13	HIGH ENERGY ASTRONOMY OBSERVATORY	Conduct long duration observations to characterize the high energy, but lower flux radiation of importance to astrophysicists and cosmologists. Locate and describe stellar sources of high energy photons and particles.	21494	11.2 x 49.2	200/200/30
5	10	SOLAR ORBIT PAIR (A) (EA)	To monitor all of the solar sphere simultaneously and to continuously provide information on flares, sun spots, solar winds, etc.	1892	10.2 x 12.1	19324/19324/30
6	11	SOLAR ORBIT PAIR (B) (EA)	To monitor all of the solar sphere simultaneously and to continuously provide information on flares, sun spots, solar winds, etc.	2508	10.2 x 12.1	1 A. U. Helio/28.5

TABLE 3.1.1.2.1 (cont'd-3)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA ASTRONOMY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
7	12	OPTICAL INTERFER-OMETER (A)	To measure stellar diameters and IR spectra. This is achieved by two spacecraft, (A) and (B), separable up to 300 meters and each with 2 ft diameter mirror	3102	6.9 x 10.2	19324/19324/30
8	12	OPTICAL INTERFER-OMETER (B)	To measure stellar diameters and IR spectra. This is achieved by two spacecraft, (A) and (B), separable up to 300 meters and each with 2 ft diameter mirror	3102	6.9 x 10.2	19324/19324/30
9	9	RADIO INTERFER-OMETER	To measure radio spectra & radio diameter and velocities of space objects. One leg of the inter-ferometry will be earth based.	10406	14.1 x 24.9	38649/38649/28.5
10	1	ASTRONOMY EXPLORER	Independent investigations of solar and stellar behavior in the UV, X-ray and radio spectral regions.	900	4.6 x 3.9	270/270/28.5

TABLE 3.1.1.2.1 (cont'd-4)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA ASTRONOMY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
11	2	ASTRONOMY EXPLORER	Independent investigations of solar and stellar behavior in the UV, X-ray and radio spectral regions.	900	4.6 x 3.9	19324/19324/0
12	6	ORBITING SOLAR OBSERVATORY	Monitor temporal variations of the Sun's brightness in the UV, X-ray and gamma-ray regions.	2000	6.9 x 10.2	350/350/28.5

TABLE 3.1.1.2.2

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA SPACE PHYSICS MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	3	LOWER MAGNETO-SPHERE	To conduct investigations of the environment of the lower magneto-sphere, neutral air chemistry and density, and ionospheric behavior.	2000	3.9 x 7.9	180/18000/90 & 28.5
2	4	MIDDLE MAGNETO-SPHERE	To measure ionospheric current systems and behavior with respect to solar activity. Also neutral atmospheric studies.	1000	5.9 x 7.9	1000/20000/90 & 28.5
3	5	UPPER MAGNETO-SPHERE	To monitor "space weather" and the boundary of the geomagnetic field as it interacts with the solar wind.	600	3.9 x 5.9	1 A.U. Helio/ Ecliptic
4	7	GENERAL RELATIVITY	To experimentally test Einstein's general relativity theory. Gyro-scopes in an earth orbiting satellite will experience two relativistic precession effects.	1500	4.9 x 6.9	300/300/?

TABLE 3.1.1.2.2 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA SPACE PHYSICS MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS.		PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
PAY-LOAD	FLEM-ING					
5	8	GENERAL RELATIVITY	To experimentally test Einstein's general relativity theory. Gyroscopes in an earth orbiting satellite will experience two relativistic precession effects.	500	3.9 x 4.9	1 A.U. Helio/28.5

TABLE 3.1.1.2.3

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA EARTH OBSERVATION MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
			R&D			
1	21	POLAR EARTH OBSERVATION SATELLITE	To design, develop and operate a space observatory system to perform meteorological and earth resources surveying by advanced remote sensing techniques	2600	12.1 x 15.1	500/500/100
2	22	SYNCHRONOUS EARTH OBSERVATION SATELLITE	Research satellite to investigate and develop remote sensing techniques for measurement of the Earth's surface and atmosphere from synchronous altitude	1000	5.9 x 3.9	19324/19324/0
3	23	EARTH PHYSICS SATELLITE	To make precision measurements of the Earth's land and sea areas to determine (1) continental drift, (2) mass distribution, (3) surface strain, and (4) variation of gravi gravity, sea altitude, and mass.	600	3.6 x 9.8	400/400/90

TABLE 3.1.1.2.3 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA EARTH OBSERVATION MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
4	27	SYNCHRO-NOUS EARTH RESOURCES SATELLITE	To design, develop and operate a satellite system for remote sensing of the Earth's surface and the lower regions of the atmosphere from synchronous orbital altitude.	1030	3.9 x 5.9	19324/19324/0
5	24	SYNCHRO-NOUS METEROLOGY SATELLITE	Develop and operate a synchronous meterological satellite for DOC/ESSA.	1030	4.9 x 7.9	19324/19324/0
6	25	TIROS	System demonstration of the 4th generation series of operational meteorological satellite for DOC/ESSA	1030	4.9 x 9.8	702/702/100
7	26	POLAR EARTH RESOURCES SATELLITE	To design, develop and operate a space observatory system to perform meteorological and Earth resources surveying by advanced remote sensing techniques	2600	11. 8 x ?	500/500/100

TABLE 3.1.1.2.4

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA COMMUNICATIONS AND NAVIGATION MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	28	APPLICATION TECHNOLOGY SATELLITE	Earth to geo-sationary orbit communication power, high gain multibeam satellite antenna, general application technology (meteorology, earth observations, etc.).	8230	15.1 x 21.0	19324/19324/0
2	30	SMALL APPLICATION TECHNOLOGY SATELLITE	To design, develop, launch, and operate a series of small R&D satellites for the experimental application of research and technology developments in spacecraft and sensor subsystems.	620	6.6 x 12.1	300/3000/90
3	29	SMALL APPLICATION TECHNOLOGY SATELLITE	To design, develop, launch, and operate a series of small R&D satellites for the experimental application of research and technology developments in spacecraft and sensor subsystems.	620	6.6 x 12.1	19324/19324/0

TABLE 3.1.1.2.4 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA COMMUNICATIONS AND NAVIGATION MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
4	31	COOPERATIVE APPLICA-TIONS	Communication satellite to be flown in partnership with other nations who will provide corre-sponding technical and funding assistance.	850	6.6 x 3.7	19324/19324/0
5	32	COOPERATIVE APPLICA-TIONS	Communication satellite to be flown in partnership with other nations who will provide corre-sponding technical and funding assistance.	850	6.6 x 12.1	300/3000/90
6	33	MEDICAL NETWORK SATELLITE	Facilitate applications of space technology and satellite systems for medical data transmission purposes.	2070	12.1 x 15.1	19324/19324/0
7	34	EDUCATION BROADCAST	Facilitate application of space technology and satellite systems for educational broadcast purposes.	3520	9.8 x 24.9	19324/19324/0

TABLE 3.1.1.2.4 (cont'd-3)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA COMMUNICATIONS AND NAVIGATION MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAYLOAD	IDENT. NOS. FLEMING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
8	35	FOLLOW-ON SYSTEMS DEMONSTRATION	Systems demonstration satellites for law enforcement, air traffic control, land traffic control type missions.	2070	12.1 x 15.1	19324/19324/0
9	36	TRACKING AND DATA RELAY	Develop and operate a command, tracking and data relay of low orbiting satellites from a synchronous satellite to a few centrally located mission control centers.	2380	9.8 x 17.1	19324/19324/0

TABLE 3.1.1.2.5

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA PLANETARY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	50	MARS VIKING	To provide information regarding the possible existence and nature of life on Mars, the atmospheric and surface characteristics of the planet and the nature of the planetary environment.	7720	9.8 x 12.1	Interpl. Helio. + Mars Plan Planeto
2	52	VENUS EXPLORER ORBITER	Measure plant magnetosphere, magnetosheath, detached bow shock-wave, and tail and wake region. Investigate internal composition, structure and magnetic field.	1000	4.9 x 12.1	Interpl. Helio. + Venus Planeto
3	53	VENUS RADAR MAPPING	Detailed surface mapping of Venus to a resolution of 50 meters using radar imaging.	7900	9.8 x 24.9	Interpl. Helio + Venus Planeto

TABLE 3.1.1.2.5 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA PLANETARY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
4	54	VENUS EXPLORER LANDER - 1	Analysis of surface properties and environment of Venus. Measurement of atmospheric properties during descent. Surface mapping by orbiter.	7420	9.8 x 24.6	Interpl. Helio + Venus Planeto
5	54	VENUS EXPLORER LANDER - 2	Orbiting microwave and IR special instruments for surface, atmosphere and cloud studies. Landed seismometer, X-ray diffraction, composition measurement, environmental dynamics.	4750	9.8 x 24.6	Interpl. Helio + Venus Planeto
6	56	GRAND TOUR	Obtain first-generation flyby data of Uranus and Neptune. Correlate spatial effects in cosmix flux and solar wind with JSP mission.	1500	9.8 x 12.1	Interpl. Helio + Jupiter Swingby + Uranus & Neptune Flyby

TABLE 3.1.1.2.5 (cont'd-3)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA PLANETARY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
7	55	JUPITER PIONEER ORBITER	Measure particles and field environment to 5AU, particle density of asteroid belt, magnetic and radiation fields of Jupiter, and to provide Jupiter imaging.	950	9.8 x 15.1	Interpl. Helio. + Jupiter Planeto
8	57	JUPITER TOPS ORBITER PROBE	Monitor particles and field environment, measure atmospheric composition, characteristics and profiles.	3290	9.8 x 15.1	Interpl. Helio. + Jupiter Planeto
9	58	URANUS TOPS ORBITER PROBE	Mapping, composition analysis, and time dependent measurements of the atmosphere. Determine the extent and intensity of planetary fields.	3700	9.8 x 15.1	Interpl. Helio. + Uranus Planeto
10	59	ASTEROID SURVEY	Define micrometeoroid, particle and field environment in asteroid belt. Prove solar electric propulsion over long duration.	1900	9.8 x 20.0	Interpl. Helio.

TABLE 3.1.1.2.5 (cont'd-4)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NASA PLANETARY MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
11	60	COMET RENDEZVOUS	Close range, long duration examination of comet. Determine physical state, structure, composition and mode of interaction with the planetary environment.	2070	9.8 x 20.0	Interpl. Helio.
12	51	MARS SURFACE SAMPLE RETURN-A	The exploration of Mars and the return of physical samples of the planetary surface to Earth. Orbiter 1 Bus vehicle.	10600	14.1 x 16.1	Interpl. Helio. + Mars Planeto
(Mated with B in Earth orbit for flight to Mars						
13	52		The exploration of Mars and the return of physical samples of the planetary surface to Earth. Lander 1 Return Probe.	11400	14.1 x 23.0	Interpl. Helio. + Mars Planeto + Earth Planeto
(Mated with A in Earth orbit for flight to Mars)						

TABLE 3.1.1.2.6

PAYLOAD CHARACTERISTICS - NASA SHUTTLE SORTIE MISSIONS 1979-1990 (MANNED)

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	38	GENERAL SCIENTIFIC RESEARCH	Provide manned research module to conduct astronomy, space physics, life science, and/or contamination monitoring experiments while attached to shuttle.	27,500	14.1 x 54.1	200/200/55
2	39	GENERAL APPLICA-TIONS	Provide research module for man to conduct earth observations, communications and navigations, and/or material science experi-ments while attached to shuttle.	30,000	14.1 x 54.1	100/100/165
3	40		Provide a dedicated research module for man to conduct Earth observations, communications and navigations, and/or material science experiments while attached to shuttle.	29,500	14.1 x 54.1	2000/200/55

TABLE 3.1.1.2.6 (cont'd-2)

PAYLOAD CHARACTERISTICS - NASA SHUTTLE SORTIE MISSIONS 1979-1990 (MANNED)

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAYLOAD	IDENT. NOS. FLEMING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
4	41	DEDICATED APPLICATIONS	Provide module for man to conduct short-term Earth observations while attached to the shuttle.	22,500	14.1 x 41.0	100/100/75
5	42	EARTH OBSERVATION	Provide a test bed to conduct scientist-astronaut training and automated experiments.	6000	14.1 x 37.1	125/125/90
6	43		Provide a test bed to conduct scientist-astronaut training and automated experiments.	4300	14.1 x 37.1	200/200/28.5
7	44	ASTRONOMY	Provide a test bed to conduct scientist-astronaut training and automated experiments.	5700	14.1 x 37.1	200/200/28.5
8	45	FLUID MANAGEMENT	Provide a test bed to conduct scientist-astronaut training and automated experiments.	7100	14.1 x 37.1	200/200/28.5

TABLE 3.1.1.2.6 (cont'd-3)

PAYLOAD CHARACTERISTICS - NASA SHUTTLE SORTIE MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
9	46	TELEOPERA-TOR	Provide a test bed to conduct scientist-astronaut training and automated experiments.	5000	14.1 x 37.1	200/200/28.5
10	47	MANNED WORK PLATFORM	Provide a test bed to conduct scientist-astronaut training and automated experiments.	6700	14.1 x 37.1	200/200/28.5
11	48	LARGE TELESCOPE MIRROR TEST	Provide a test bed to conduct scientist-astronaut training and automated experiments.	13,000	14.1 x 54.1	200/200/28.5
12	49	ASTRONAUT MANEUVERING UNIT (AMU)	Provide a test bed to conduct scientist-astronaut training and automated experiments.	3800	14.1 x 37.1	200/200/28.5

TABLE 3.1.1.2.7

PAYLOAD CHARACTERISTICS - NASA SPACE STATION MODULES 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	61	CORE MODULE	Long-term manned space operations, core module.	20,000	14.1 x 39.4	270/270/55
2	62	POWER MODULE	Long-term manned space operations, power module.	20,000	14.1 x 30.2	200/200/55
3	63	CREW MODULE	Long-term manned space operations, crew module.	20,000	14.1 x 30.2	200/200/55
4	64	CONTROL MODULE	Long-term manned space operations, control module.	20,700	14.1 x 30.2	200/200/55
5	65	GENERAL PURPOSE LABORATORY	Long-term manned space operations, general purpose laboratory.	20,700	14.1 x 30.2	200/200/55
6		MIN-MOD BIG GEMINI CREW MODULE	Resupply of orbiting space station and transporation of 9 men, crew module.	23,550	15 x 49.2	200/200/55

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TABLE 3.1.1.2.7 (cont'd-2)

PAYLOAD CHARACTERISTICS - NASA SPACE STATION MODULES 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
7		MIN-MOD BIG GEMINI-CARGO PROPULSION MODULE	Resupply of orbiting space station and transportation of 9 men, cargo/propulsion module.	21,760	15 x 24.9	200/200/55
8		ADVANCED BIG GEMINI-CREW MODULE	Resupply of orbiting space station and transporation of 12 men, crew module.	24,420	21.6 x 52.5	200/200/55
9		ADVANCED BIG GEMINI-CREW MODULE	Resupply of orbiting space station and transportation of 12 men, cargo/propulsion module.	104,690	21.6 x 42.6	200/200/55
10	66	EXP. MOD. -1 LIFE SCIENCE	Support of space biology experi-ments.	27,600	15 x 58.1	200/200/55

TABLE 3.1.1.2.7 (cont'd-3)

PAYLOAD CHARACTERISTICS - NASA SPACE STATION MODULES 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
11	67	EXP. MOD. -1 EARTH OBS.	Support of earth observation experiments.	27,600	15 x 58.1	200/200/55
12	68	EXP. MOD. -1 SPACE MFG.	Support of material science and processing experiments.	27,600	15 x 58.1	200/200/55
13	64	EXP. MOD. -3 PHYS. LAB	Support of space physics experiments.	24,880	15 x 41.0	200/200/55
14	65	EXP. MOD -3 COSMIC RAY LAB.	Support of cosmic ray experiments.	24,880	15 x 41.0	200/200/55
15	68	EXP. MOD. -3 COMMN. & NAVIG.	Support of communication and navigation experiments.	24,880	15 x 41.0	200/200/55

TABLE 3.1.1.2.8

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NON-NASA OPERATIONAL MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
1	70	COMMUNICATION SATELLITE	Provide operational services in information networks and navig.	1490	8.9 x 22.0	19304/19304/0
2	71	US DOMESTIC COM. SATELLITE	Provide operational services in communication networks, cable TV, broadcast TV, radio, telephone, teletype, etc.	3540	15.1 x 24.9	19304/19304/0
3	72	FOREIGN DOMESTIC COMM. SATELLITE	Provide operational services in comm. networks for S. Amer., Can., Australia, ESRO, S. Africa, India, and neighboring countries.	1030	3.9 x 12.1	19304/19304/0
4	73	NAVIG./ TRAFFIC CONTROL SATELLITE	To gather data from remote mobile platforms and scattered transmitters and centralize the outputs into a common data control.	730	4.9 x 7.9	19304/19304/5

TABLE 3.1.1.2.8 (cont'd-2)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NON-NASA OPERATIONAL MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
5	74	NAVIG./ TRAFFIC CONTROL SATELLITE	Navig. data over oceans and domestic areas.	730	4.9 x 7.9	15984/29970/29
6	75	TOS METEOR-OLOGICAL SATELLITE	Observe global clouds, day and night, cloud top heights, heat balance, vertical temp. and water vapor profiles.	1030	4.9 x 5.9	700/700/100
7	76	SYNCHRO-NOUS ME-TEOR SAT.	Operational meteorological satel-lite operating from synchronous alt. for ESSA.	1030	4.9 x 7.9	19303/19303/0
8	77	POLAR EARTH RESOURCES	Operational satellite to continu-ally survey Earth resources and to perform meteorological survey with high resolution sensors and trans-mit data to Earth.	2590	5.9 x 5.9	500/500/100

TABLE 3.1.1.2.8 (cont'd-3)

CURRENT EXPENDABLE PAYLOAD CHARACTERISTICS - NON-NASA OPERATIONAL MISSIONS 1979-1990

("FLEMING" MODEL, SPRING 1971)

IDENT. NOS. PAY-LOAD	IDENT. NOS. FLEM-ING	PAYLOAD	MISSION	LAUNCH MASS, LB.	DIA & LGTH, FT. X FT.	ORBIT PERI/APO/INCL. N.M./N.M./DEG.
9	78	SYNCH-RONOUS EARTH RESOURCES	Operational remote sensing and measurement of the Earth's resources and lower atmosphere.	1030	5.9 x 5.9	19304/19304/0
10	78	SYNCH-RONOUS EARTH RESOURCES	Operational remote sensing and measurement of the Earth's resources and lower atmosphere.	1030	5.9 x 5.9	19304/19304/0

3.1.1.3 Solid Rocket Motors

3.1.1.3.1 156" Booster SRM

Each of the two SRM's is 156 inches in diameter and 1802 inches long, nose to nozzle end. Each motor contains six segments of solid propellent, 6 separation solid rocket motors and two thrust termination ports forward. The nozzles have a
168 fixed inboard cant angle of 11°. The nose cone of each motor contains the ribbon type parachute and recovery tracking electronics. Mounting attachment points are described under 3.1.1.4. With respect to the External Tank coordinate system the motors are located as follows:

156" SRM	X	Y	Z
LH Motor	919 to 2721	-245	448
RH Motor	919 to 2721	+245	448

The nose cone contains 3 110 ft. diameter ribbon type chutes, reefing cutters, flotation bag, battery and a flashing light with salt water switch. Two antennas are located on the Z axis 180° apart near X station 1133. The forward end of the rocket also contains:

2 receivers
2 decoders
2 sequencers
2 safe/arm units
4 batteries

3.1.1.3.2 SRM Separation Rockets

Each of the launch boost SRM's contain 3 separation rockets forward and 3 aft. The vacuum thrust of each separation rocket
178 is 27,000 lbs. with a burn time of 2 seconds. With respect
pp 3-3 to the boost SRM X axis, the forward thrusters are canted aft 45° and the aft thrusters are canted forward 45°. The forward thrusters are rolled 20° off vertical, the aft 40° off vertical.

3.1.1.3.3 Abort SRM's

The Abort SRM's are located inboard atop each orbiter wing attached to the orbiter fuselage. Each rocket is 300 inches
166 long and 65.7 inches in diameter. Approximately 30 seconds
pp 3-81 after liftoff, the ASRM's are jettisoned and are not recovered. The nozzles have a minus 3° cant in the pitch plane to allow maximum elevon travel. Thrust is aligned through the orbiter center of gravity. Interface to the orbiter consists of electrical safe, arm and igniter firing. The ASRM's are separated by explosive bolts and a separation ramp. Mounting coordinates with respect to the orbiter coordinate system are:

	X	Y	Z
ASRM	1307 to 1550	168	352

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pp 2-4

3.1.1.3.4 ET Deorbit SRM

The External Tank deorbit SRM is housed in the nose section
171 of the tank. Orientation of the motor with respect to the
tank is fixed to align the thrust vector with the tank x axis.
pp 2-59, 2-60, 2-66 The motor dimensions are:

Length = 75.3 inches including nozzle
Diameter = 37 inches

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	X	Y	Z
ASRM	1307 to 1550	168	352

167 pp 2-4

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tank is fixed to align the thrust vector with the tank x axis.
pp 2-59, 2-60, 2-66 The motor dimensions are:

Length = 75.3 inches including nozzle
Diameter = 37 inches

3.1.1.4 External Tank

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pp 2-59, 60, 61

The External Tank is cylindrical in shape tapering at a 20° angle to a cylindrical housing cover for the deorbit solid rocket motor. The tank has 3 attachment points to the orbiter vehicle and 4 to each SRM. The coordinate system is centered 200 inches forward of the nose of the vehicle with plus X measured toward the Tank, + Y measured parallel to the orbiter, RH wing and +Z measured toward the orbiter.

	X	Y	Z
Orbiter forward	1572	0	+180.5
Orbiter aft	2326	±123.0	+186.25
SRM forward	1133	±162.5	0
SRM aft (center slide)	2388	±159.4	31.2
SRM aft (upper)	2388	± 86.0	+141.0
SRM aft (lower)	2388	±130.0	- 52.0

The two orbiter aft attach points include the umbillicals (Propulsion fluid lines, electrical harness). The SRM electrical interface is through the aft upper and lower attach points. The cylindrical nose housing for the deorbit SRM is 41.0 inches diameter and 1240 inches long. The ellipsoidal dome LO_2 tank has a volume of 21246 ft^3 and is located aft of the SRM (station 324) to station 1066. The LH_2 tank has a volume of 58631 ft^3 and located between stations 1081 and 2440. Range safety equipment consisting of:

4 Electrical power batteries
2 Safe/arm units
2 Sequencers
2 Decoders
2 Receivers
4 Antennas

are located in the area between the LO_2 and LH_2 tanks. The antennas locations are approximately:

	X	Y	Z
#1	1026	+115	+115
#2	1026	+115	-115
#3	1026	-115	-115
#4	1026	-115	+115

Point sensors are located in each tank for propellant utilization guaging. Each tank contains baffling to reduce slosh. All fluid controls and valves for the ET operation are located within the Orbiter. The ET instrumentation consists of the transducers required for pressure, temperature, strain and vibration data.

The External Tank outer dimensions are:

Length = 2067 inches
Diameter = 318 inches

3.1.1.5 Air Breathing Engines

The operational space mission ABPS engines are installed as a module in the aft section of the cargo bay. Two engines are
166 installed in the module located as follows (referenced to Orbiter coordinates):

DEPLOYED	X	Y	Z
RH Engine	1068 to 1302	-70.5	532.5
LH Engine	1068 to 1302	+70.5	532.5

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pp 2-4

STOWED	X	Y	Z
RH Engine	1068 to 1302	-	464.223
LH Engine	1068 to 1302	-	464.223
Fuel Tank:	1113	0	358

Fuel tank capacity = 21,000 lbs.

The coordinates are with respect to the Orbiter reference system.

3.1.1.6 OMS Engines

Two OMS engines are installed in the aft end of the fuselage. In the orbiter coordinate system:

	X	Y	Z
OMS Gimbal	1547	± 96	454

The OMS yaw null is 0° with gimbal capability of 0° to 12° outboard. The pitch null is 11.5° with $\pm 4°$ gimbal capability.

3.1.1.7 RCS Engines

The RCS is contained in three modules, one integrated in each OMS pod and one in the orbiter fuselage nose. Each aft pod contains 12 thrusters; the forward module contains 16 thrusters.

3.1.1.8 Docked Configuration

The docking tunnel is located overhead behind the commander/ pilot stations centered at stations:

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X = 617

Y = 0

Z = 520

above the 78 inch outer diameter airlock. The docking tunnel retracts into the airlock for stowage. A front panel of the tunnel opens upward providing a 40 inch clearance for EVA. Payloads will contain a radial docking port and alignment

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pp 17

accomplished with an overhead window COAS at the cargo manipulator station. Docking with another orbiter is assumed to be port to port. The extended tunnel provides a 36 inch

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pp 3-53

clearance between the docking element and orbiters.

3.1.2 Operational Ferry Mission Configuration

The ABES module containing two engines is being replaced with a module containing four engines. Additional fuel is supplied by a tank loaded in the cargo bay forward of the Mission ABPS tank. The geometry is:

	X	Y	Z
Engine 1	1068 to 1302	136.495	464.223
Engine 2	1068 to 1302	70.5	532.5
Engine 3	1068 to 1302	-70.5	532.5
Engine 4	1068 to 1302	-136.495	464.223
Ferry Tank	800 to 1000	0	371.827

The ferry mission tank contains 26,000 lbs. of fuel. The mission tank remains installed for ferry missions and has a capacity of 21,000 lbs. of fuel.

3.1.3 Horizontal Flight Test

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The horizontal flight test configuration includes four airbreathing engines installed as a module in the aft section of the payload bay, additional JP fuel tankage, smooth fairings replacing the OMS/RCS pods and ejection seats for the flight crew. The flight crew will consist of the commander and co-pilot only. The equipment required solely for space operation will be omitted for Orbiter 1 but will be included in Orbiter 2.

3.1.4 Vertical Flight Test

Orbiter 2 will be used with Development Flight Instrumentation (DFI). The DFI will be removed from Orbiter 1 after horizontal flight tests are complete and from Orbiter 2 after a planned 14th orbital flight. The first vertical flight will be unmanned with chase planes and/or ground control backup to the auto systems. The vehicle will be configured for a once-around with landing at Edwards AFB or a 2 1/2 day mission. Ejection seats will be installed for early flights. The first flight, if unmanned, will include a GN & C subsystem sequence controller, and an S-band TV camera system will be installed.

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pg. 7-11

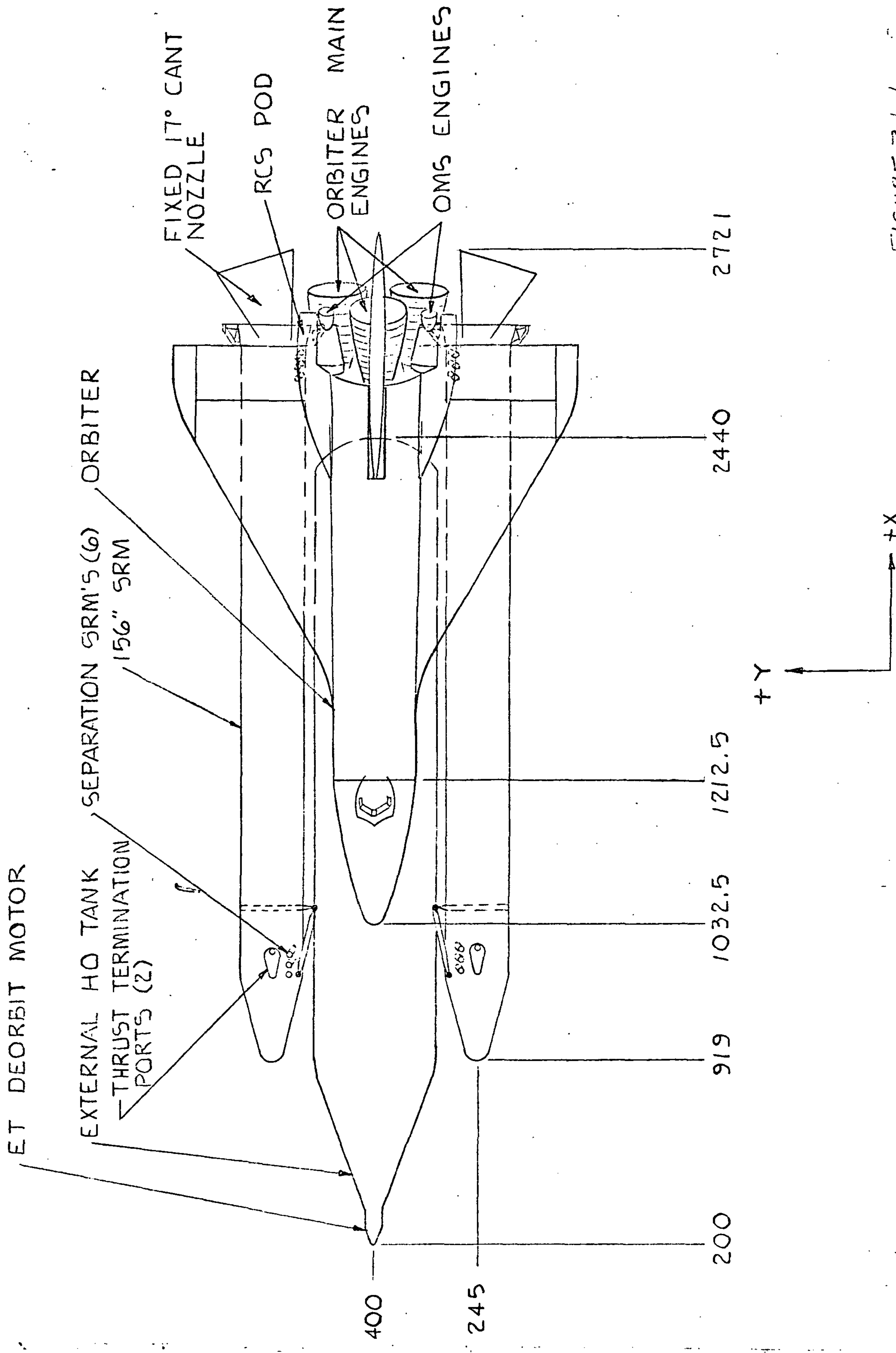

ET DEORBIT MOTOR
EXTERNAL HO TANK
THRUST TERMINATION PORTS (2)
SEPARATION SRM'S (6)
156" SRM
ORBITER
FIXED 17° CANT NOZZLE
RCS POD
ORBITER MAIN ENGINES
OMS ENGINES
2721
2440
1212.5
1032.5
919
200
400
245
+Y
+X

FIGURE 3.1-1

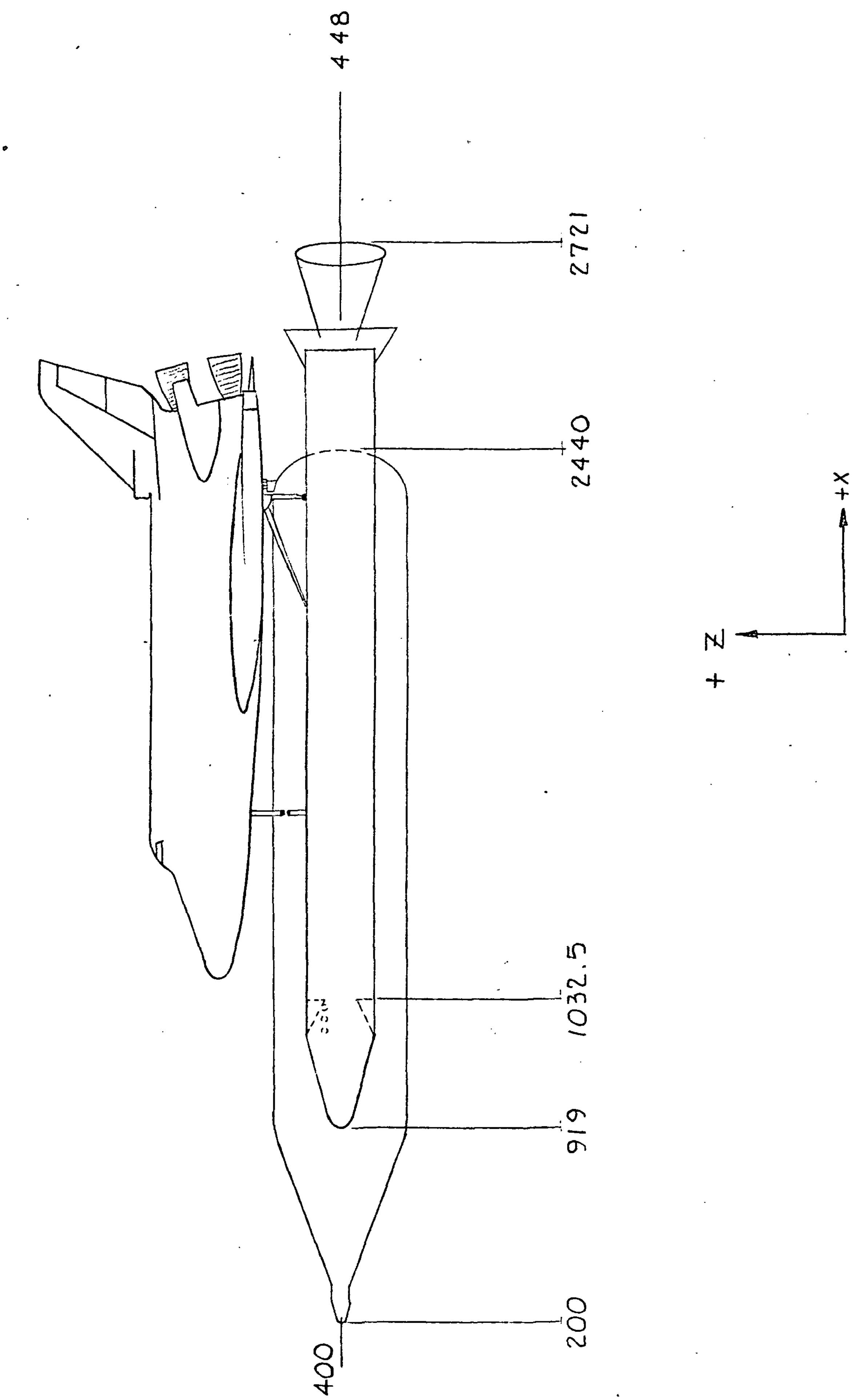
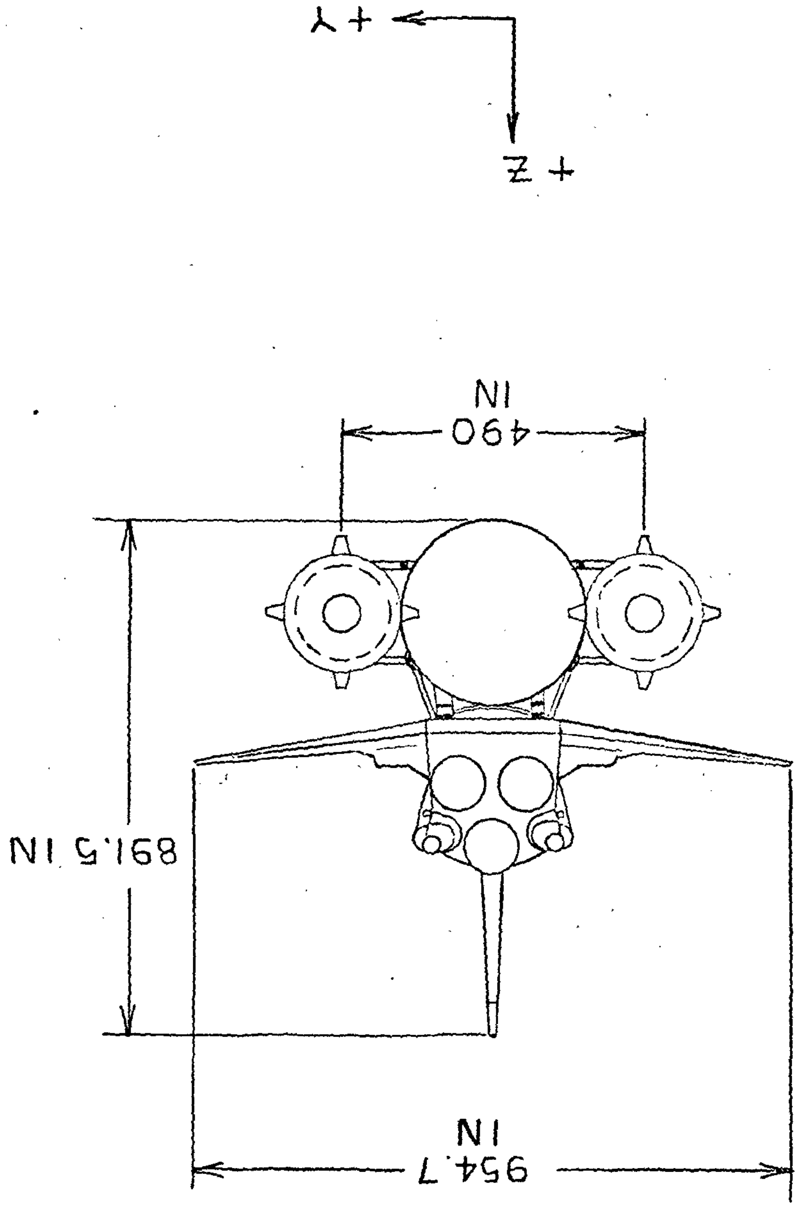


FIGURE 3.1-2


+Y
+Z
490 IN
891.5 IN
954.7 IN

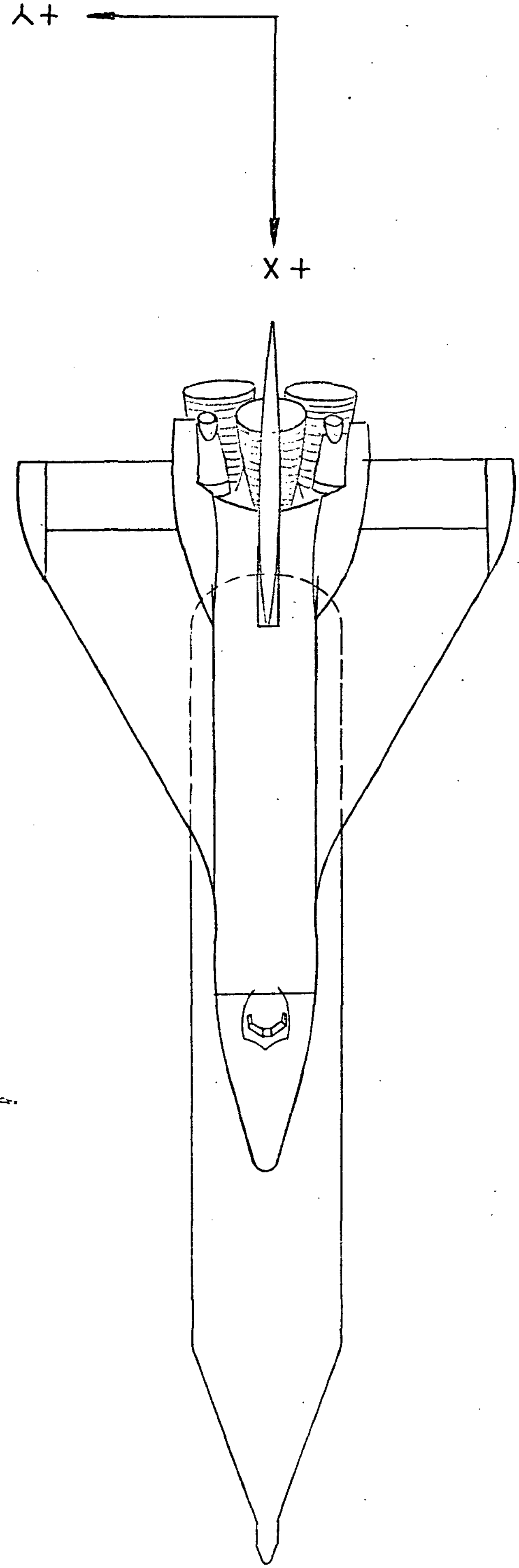


FIGURE 3.1-4

FIGURE 3.1-5

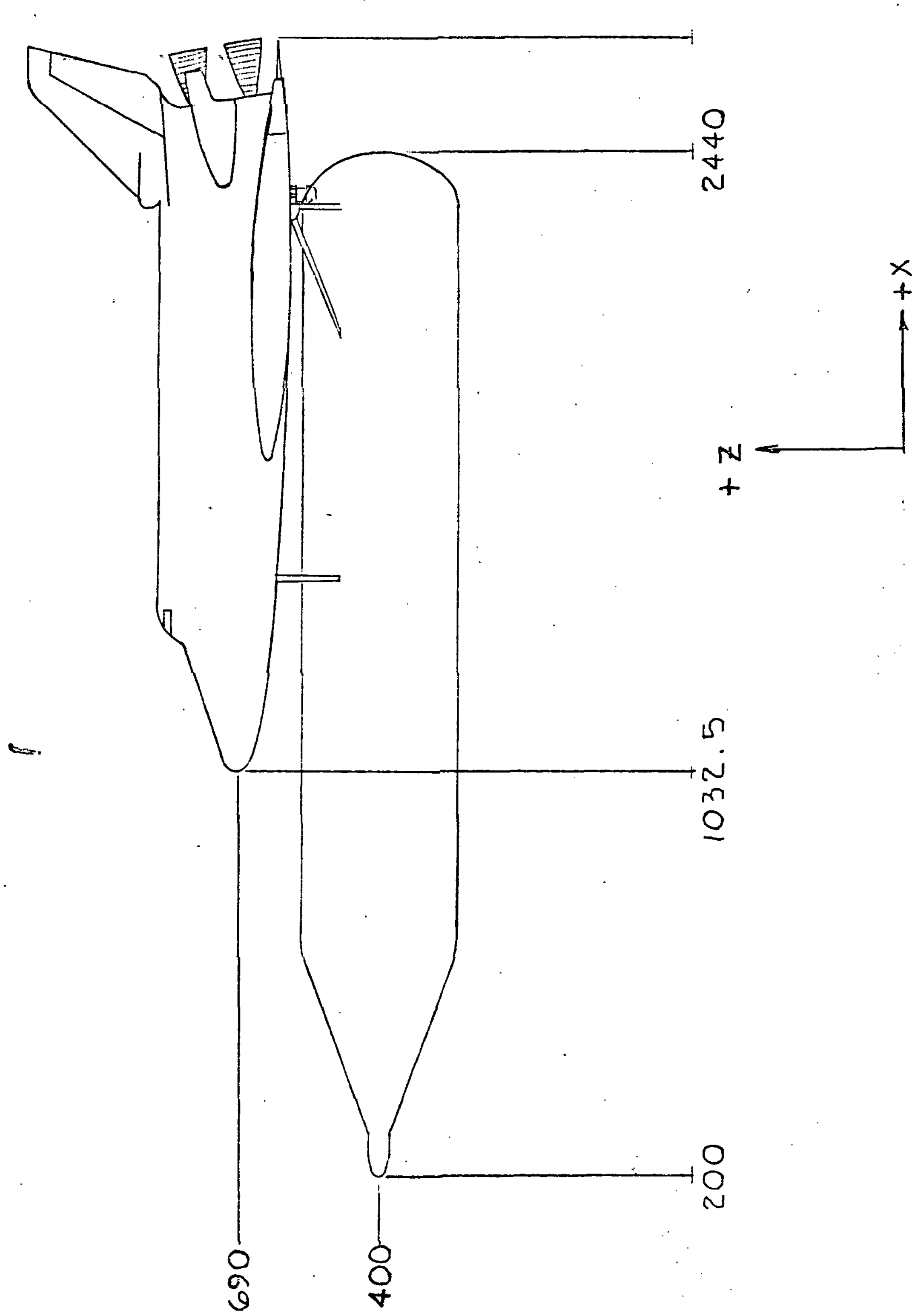
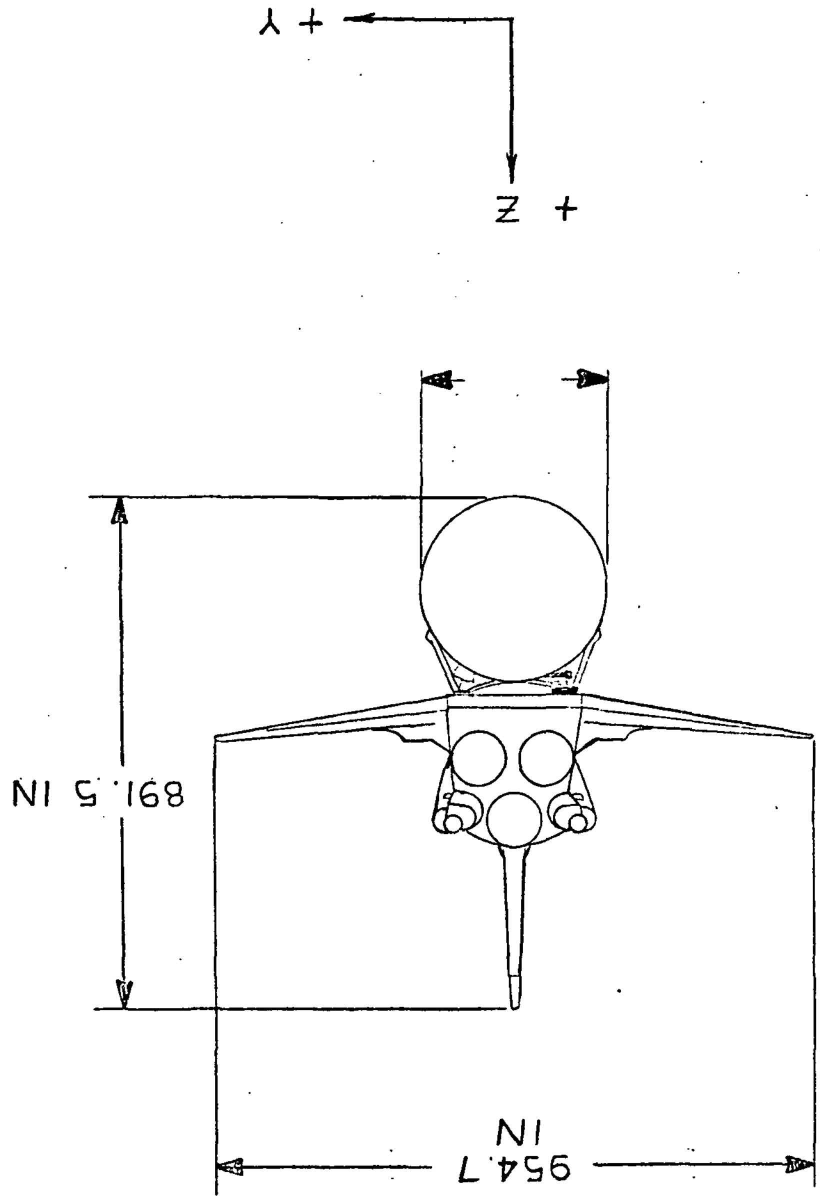


FIGURE 3.1-6



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p.2-5,
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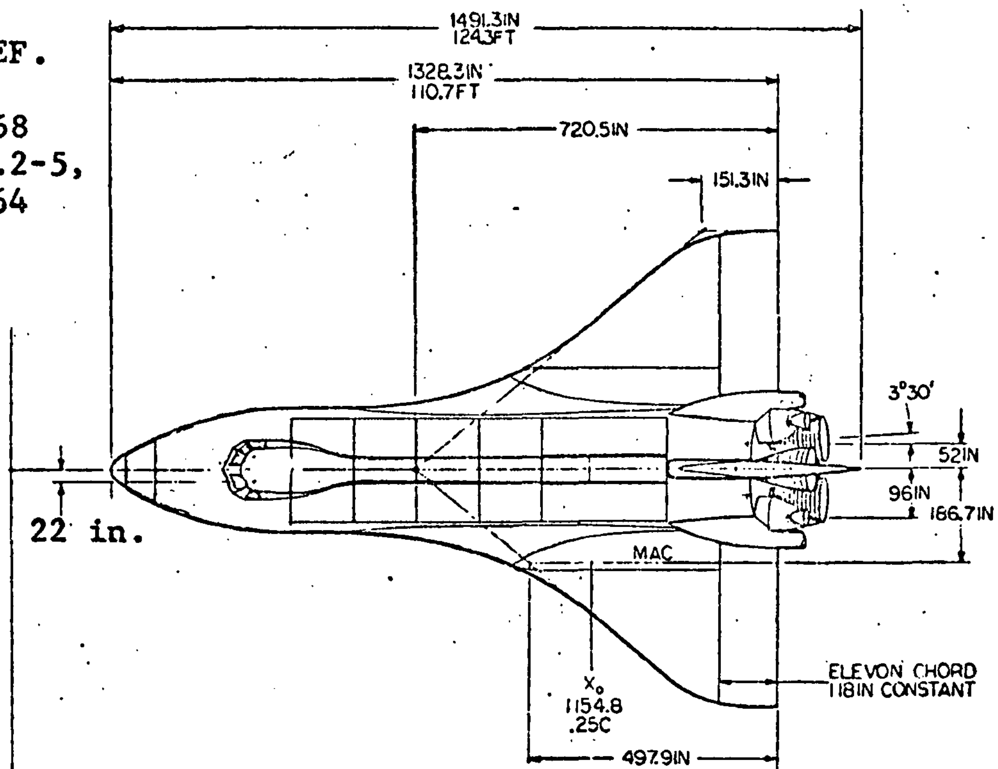

GEOMETRY	WING	VERTICAL TAIL
AREA-FT^2	2890	402
ASPECT RATIO	2.19	1.68
TAPER RATIO	.21	.40
INCIDENCE ANGLE	+3° AT Y_o125	—
	−2° AT Y_o477.4	—
AIRFOIL	0008 64 AT Y_o125	10° SYMMETRICAL
	2412 AT Y_o429.6	60-40 WEDGE
	BOTH MODIFIED	

LANDING GEAR	MAIN	AUXILIARY
TIRE SIZE	44.5X16-21	32X8.8
TIRE TYPE	NR/USAF B-1	VII
PLY RATING	26	16
ROLLING RADIUS-IN	18.4	13.3
FLAT RADIUS-IN	13.6	10.9
STROKE-IN	16	18

	MAIN	OMS	RCS	ABE
THRUST-VACUUM-LBS	470K	5K	1K	CLASSIFIED
PROPELLANT TYPE	H_2/O_2	N_2O_4/MMH	N_2H_4	JP4
NUMBER	3	2	40	2 ORBIT 4 FERRY

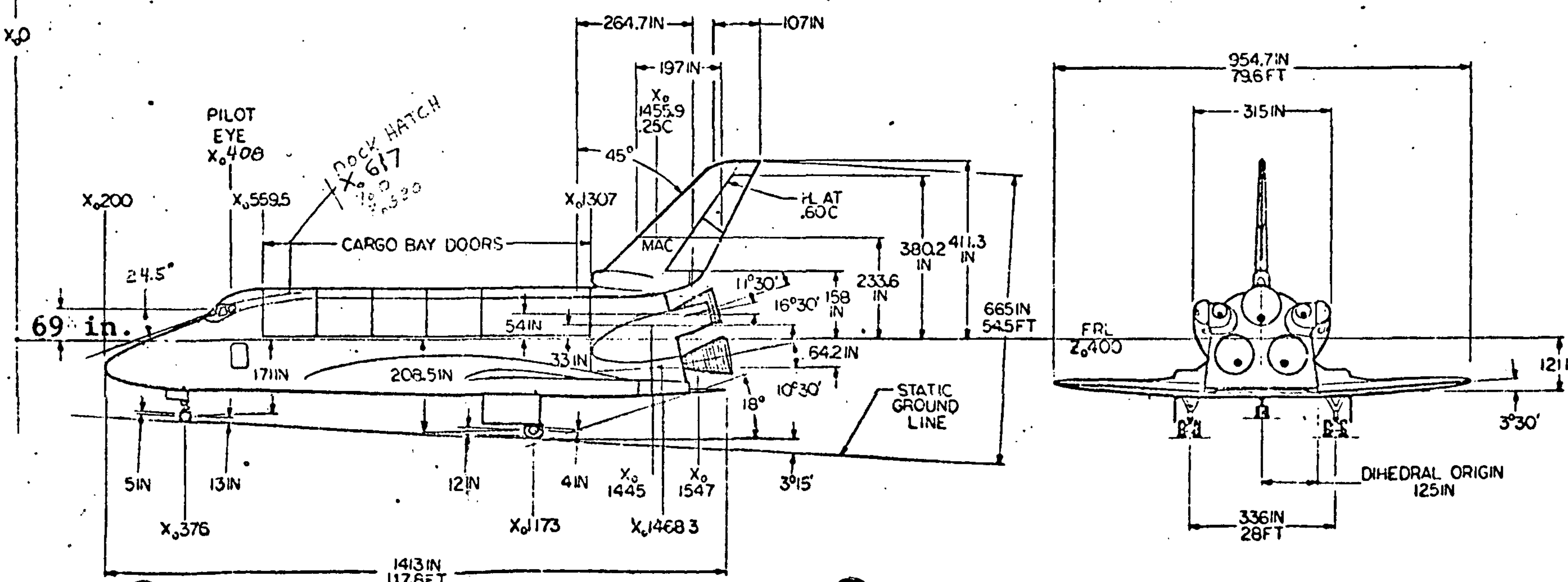

FIGURE 3.1-7

3.1.6 Data References

Document #12 Pages 3-41, 3-42

Document #15 Page II-31

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Document #166

Document #167

Document #164

Document #168 Pages 2-5

REF. KEY

3.2 Equation of Motion

3.2.1 Coordinate Systems

3.2.1.1 Requirements

Simulation of Space Shuttle Missions will require a variety of coordinate systems. Several systems or classes of systems for which a requirement is anticipated are listed below:

1. True-of-date Inertial Besselian, Earth-Centered (or sufficiently good approximation of true-of-date)

This system should be the basic EØM reference coordinate system for all flight regimes except, possibly, approach and landing. (It should not be used as basic system for approach/landing unless all calculations are accomplished in at least 32 bit-mantissa floating point - even 32 bits is marginal.) If accuracy requirements are very high, the system may have to be updated occasionally to account for precession and nutation. An example system would be:

x-axis: intersection of true equator and vernal equinox on date

z-axis: earth true north-polar spin axis on date

y-axis: completes triad

2. Ephemeral Inertial Besselian, Earth-Centered

This system's definition will probably be determined by the system in which the available ephemeris data is expressed. If accuracy requirements are low enough to permit precession effects to be ignored over the time-span of interest, this system could be identical to system 1. (If supplied data was not in this coordinate system, off-line transformation would be necessary to put it in the system). The positions of stars, planets, moon, and sun will probable be expressed in this system. An example system would be (mean 1950):

REF. KEY

x-axis: intersection of mean equator of 1950 and vernal equinox

z-axis: earth mean north-polar spin axis of 1950

y-axis: completes triad

3. Geographic Earth-Fixed, Earth-Centered

Several such systems may be required. Location of tracking stations, TACAN transmitters, landing sites, etc., will be expressed in such systems. An example system would be:

x-axis: intersection of instantaneous true equator and Greenwich meridian

z-axis: earth north-polar instantaneous true spin axis

y-axis: completes triad

4. Landing Field Earth-Fixed, Landing Field-Centered

This system may be the basic EØM reference system during approach and landing. In any case, it will probably be advisable to have vehicle state available within it for use in various systems (visual, gear dynamics, radar altimeter, landing aids, etc.). A sample system would be:

center: end of runway over which final approach trajectory passes, along runway center-line

x-axis: along runway center-line, opposite to landing velocity vector

z-axis: normal to reference ellipsoid at coordinate system center, positive up

y-axis: completes triad

4a. Other Earth-Fixed, point-on-earth Surface Centered coordinate systems (analogous to 4) may be advisable (e.g., systems centered at TACAN stations).

REF. KEY

5. Stable Member Coordinate System

The current alignment of the platform stable member(s). Inertial accelerometer readings are output in this system. The system is approximately inertially fixed (ignoring drift, it is inertially fixed) except during platform realignment. The exact configuration of the system is dependent on the current alignment (plus drift). Several possible alignments are:

Launch Plumbline:

x-axis: along normal to reference ellipsoid at launch site, positive up

z-axis: along launch azimuth, positive down range

y-axis: completes triad

(NOTE: It may also be necessary to make EØM state available in this system during boost for trajectory verification)

Preferred Alignment for burn at time t:

x-axis: along vehicle burn-attitude longitudinal axis, positive forward

y-axis: perpendicular to x-axis, and perpendicular to vehicle radius vector at time t

z-axis: completes triad

Local Vertical Alignment at time t:

z-axis: along line connecting vehicle and center of earth at time t, positive down

y-axis: normal to orbital plane at time t, positive in opposite sense from angular momentum

x-axis: completes triad (local horizontal)

REF.
KEY

6. Body Axis Coordinate System:

Propulsion and aero forces and moments will be computed in this system, or a closely related system. Mass properties will be maintained in this system or a system parallel to it. The system(s) will be fixed to the vehicle. An example system would be:

center: vehicle c.g.

x-axis: in plane of vehicle symmetry, positive forward in the longitudinal direction

z-axis: in plane of vehicle symmetry, positive down through vehicle base

y-axis: completes triad

7. Target Vehicle (or Payload) Body Axis Coordinate Systems (3)

These systems will be fixed to the two target vehicle bodies, propulsion and aero forces and moments on the target vehicles will be computed in these systems, centered at the target vehicle c.g.'s.

8. Manipulator Wrist TV Axes

This coordinate system will define wrist TV camera orientation, and attached payload position. It will also define (except for a roll rotation) the orientation of the attached payload.

REF. KEY

A number of transformation matrices must be retained to transfer data between coordinate systems. Several such transformations are cited below, together with frequency of recalculation requirements, information necessary to refigure them, and some of their uses.

1. Ephemeral Besselian to True-of-Date Besselian (system 2. to system 1.)

Calculation:

Frequency: Depends upon EØM accuracy requirements (and the time range over which simulation will operate). May not require recalculation at all. May require recalculation as often as once every 15 minutes. Error to be controlled is misalignment of earth spin axis and equator due to precession/nutation, which feeds into gravity and any other latitude/longitude dependent parameters. Approximate maximum precession misalignments in earth-surface position resulting from recalculation intervals of 1 year, 1 day, and 20 minutes are shown below:

	1 year	1 day	15 minutes
maximum misalignment	3000 feet	8 feet	1 inch

These errors are "steady-state," and not oscillatory.

Required Information: Reference time in some absolute time reference system (e.g., Julian centuries past epoch 1950.0)

Use: Ephemeris information will be in the Ephemeral Besselian system, and must be transformed to true-of-date Besselian for use with EØM state (or EØM state transformed to Ephemeris Besselian). May also be useful in the process of updating the true-of date system at the time the transformation is recalculated (if it is recalculated).

REF. KEY

2. Geographic Earth-Fixed to True-of-Date Besselian (system 3 to system 9)

Calculation:

Frequency: Should be recalculated at the same rate as translational EØM recalculates state information.

Required Information: Current true Greenwich hour angle.

Use: All earth-fixed locations (navaid stations, landing strips, tracking stations, terrain, earth-surface features) will ordinarily be in this system, and must be transformed to true-of-date Besselian to be used in conjenction with vehicle state (or EØM state must be also available in this system). Gravitational force must be calculated in this system. Ground track information requires use of this matrix. All other earth-fixed system transformations can be obtained by post-multiplying by a given constant matrix, and possibly translating origin in addition. Because of its simplicity, this transformation may be used implicitly rather than explicitly in some calculations (e.g., gravity).

3. Landing Field Coordinates to True-of-Date Besselian

Calculation:

Frequency: Should be recalculated at the same rate as translational EØM recalculates state information, during approach and landing, if landing field coordinates not used as prime EØM system at that time. If prime, must be calculated once at time of translation of EØM to landing field coordinates.

Required Information: Transformation matrix 2 plus position (e.g. latitude, longitude, altitude above reference ellipsoid, runway azimuth) of landing strip.

REF. KEY

Use: Landing coordinates may be prime EØM system during landing. In this case, the transformation will be needed at time of transfer of coordinate base. If not the prime EØM system, since state will be maintained therein for approach and landing, the transformation will be used to accomplish this.

4. Body to Stable Member Transformation

Calculation:

Frequency: The transformation should be calculated at the same rate as rotational EØM makes available updated body rates.

Required Information: Body rates and platform drift rates. During platform realignments, gimbal angle movements since last iteration (or, equivalently, rates due to pulsing) are also required.

Use: The outputs of the inertial accelerometers must be obtained from the body accelerations using this transformation. (NOTE: This transformation will probably be physically incorporated in the IMU simulation.)

5. Body Axis to Basic EØM Reference System (True-of-Date Besselian or landing field)

Calculation:

Frequency: Each rotational EØM cycle.

Required Information: Body axis rotational rate (or orientation quaternions/Euler angles).

Use: This transformation is required to transform propulsion and aero forces to the basic reference system for use in Translational EØM. Its inverse is required to obtain relative wind, required in body axes for Aero, from inertial velocity in the Besselian system.

REF. KEY

6. Target vehicle body axes to basic EØM reference system.

Calculation:

Frequency: Each target vehicle rotational EØM cycle.

Required Information: Target vehicle body axis rotational rates (or orientation quaternions/Euler angles), when target vehicle is a payload being moved by the manipulator, the transformation can be found from transformations 5 and 7, the manipulator wrist roll angle, the point of attachment of the manipulator arm terminal device, and the payload attitude with respect to the terminal device.

Use: Same as transformation 5.

7. Manipulator Wrist TV Axes to Shuttle Body Axes

Calculation:

Frequency: The same frequency as calculation of payload dynamics - probably also the same frequency as rotational EØM, whenever manipulator is operating.

Required Information: Manipulator joint angles (except wrist roll).

Use: The transformation will be used to describe the orientation of the wrist TV camera on the manipulator arm. It may also be used to determine the orientation of the grasped payload.

3.2.1.2 Rationale for Assumptions

Not applicable

3.2.1.3 References

Not applicable

3.2.2.1.1 Earth Gravitational

3.2.2.1.1.1 Requirements

The acceleration on the shuttle vehicle due to the Earth's gravitational field must be simulated. Inverse-square gravitational perturbation due to the non-spherical shape and non-homogeneous makeup of the earth affect the shuttle trajectory. For purposes of training simulation, earth gravitational potential is usually modeled using potential functions containing up to four perturbation terms. Three of these terms (J_2, J_3, J_4 terms; or second, third, and fourth harmonics) are in magnitude functions of vehicle altitude and latitude, the fourth (J_{22} term) is in magnitude a function of altitude, latitude, and longitude. Magnitude of perturbing accelerations due to the J_2, J_3, and J_4 terms at various latitudes in a 100 nm circular orbit are as follows:

	0°	30°	45°	60°	90°
J_2	$.047 \frac{ft}{sec^2}$	$.042 \frac{ft}{sec^2}$	$.052 \frac{ft}{sec^2}$	$.071 \frac{ft}{sec^2}$	$.093 \frac{ft}{sec^2}$
J_3	$.11 \times 10^{-3} \frac{ft}{sec^2}$	$.13 \times 10^{-3} \frac{ft}{sec^2}$	$.12 \times 10^{-3} \frac{ft}{sec^2}$	$.17 \times 10^{-3} \frac{ft}{sec^2}$	$.29 \times 10^{-3} \frac{ft}{sec^2}$
J_4	$.080 \times 10^{-3} \frac{ft}{sec^2}$	$.085 \times 10^{-3} \frac{ft}{sec^2}$	$.091 \times 10^{-3} \frac{ft}{sec^2}$	$.10 \times 10^{-3} \frac{ft}{sec^2}$	$.21 \times 10^{-3} \frac{ft}{sec^2}$

Magnitude of perturbing accelerations due to the J_{22} term at various latitudes and longitudes where maximum and minimum in a 100 m circular orbit are as follows:

LONGITUDE \ LATITUDE	0°	30°	45°	60°
-14.8°	$.47 \times 10^{-3} \frac{ft}{sec^2}$	$.37 \times 10^{-3} \frac{ft}{sec^2}$	$.28 \times 10^{-3} \frac{ft}{sec^2}$	$.18 \times 10^{-3} \frac{ft}{sec^2}$
-59.8°	$.31 \times 10^{-3} \frac{ft}{sec^2}$	$.27 \times 10^{-3} \frac{ft}{sec^2}$	$.22 \times 10^{-3} \frac{ft}{sec^2}$	$.16 \times 10^{-3} \frac{ft}{sec^2}$

Errors resulting from ignoring the J_2 harmonic are clearly serious during any flight phase. A position error of over 100 feet can build up in as little as a minute. It should be included in all gravitational calculations. The effects of the J_3, J_4, and J_{22} term are much smaller, but are still significant. Over the period of a full orbit, since the perturbative acceleration direction turns with the orbit, the perturbations cancel to a substantial extent. However, in half an orbit (2700 seconds), neglecting the J_3, J_4, and J_{22} perturbations could cause position errors of the order of magnitude of 500 feet. Thus, it would be highly desirable to include J_3, J_4, and J_{22} terms during orbital coast. During powered flight and entry, these terms (but not J_2) might be safely neglected. However, it has been found that, during a 46 minute entry, the J_{22} term can affect touchdown point by as much as 2.5 km. **It would be desirable to include all four perturbation terms during all flight phases.**

3.2.2.1.1.2 Rationale for Assumptions

Not applicable.

3.2.2.1.1.3 References

1. 4 pp. 9.12A-25

3.2.2.1.2 Other Celestial Body Gravitational

3.2.2.1.2.1 Requirements

The shuttle vehicle will be acted upon by gravitational forces from other celestial bodies besides the earth. Lunar, solar, and planetary gravitational fields will cause perturbative accelerations upon the shuttle vehicle. Largest perturbative accelerations which could be expected from several such bodies in "worst case" conditions (shuttle in 500 nm orbit, celestial body at closest approach to earth) are as follows:

CENTRAL BODY	MAXIMUM PERTURBING ACCELERATION
MOON	$5 \times 10^{-6} \frac{ft}{sec^2}$
SUN	$2 \times 10^{-6} \frac{ft}{sec^2}$
VENUS	$2.5 \times 10^{-10} \frac{ft}{sec^2}$
JUPITER	$3 \times 10^{-11} \frac{ft}{sec^2}$

In evaluating the significance of these results, it should be noted that at opposite points in an orbit, these perturbations are in approximately opposite directions. So, over a full orbit, they largely cancel. A 500 nm circular orbit takes some 6000 sec. Thus, at no point in an orbit could planetary perturbations result in a position error of as much as a tenth of an inch. They can safely be ignored. Solar and lunar perturbations are far more significant. Ignoring lunar perturbation could result in errors of the order of magnitude of 10 feet at some point in an orbit. Therefore, it would be desirable to include lunar and solar perturbations in the calculations of forces acting upon the shuttle while in orbit.

3.2.2.1.2.2 Rationale for Assumptions

Not applicable.

3.2.2.1.2.3 References

Not applicable.

3.2.2.1.3 Dynamic Body Forces

3.2.2.1.3.1 Requirements

The Translational Equations of Motion for a simulated shuttle vehicle must incorporate aerodynamic forces throughout most, if not all, of the anticipated flight envelope (altitude $\leq$ 500 nm) into orbiter trajectory calculations. At very low altitudes and on the ground, ground effects and forces must be included in the calculations of total forces on the vehicle. Force exerted by the pad upon the vehicle must be picked up before launch (or, alternately, force calculation bypassed, and inertial state updated regularly to account for earth rotation). Before horizontal takeoff and after touchdown, normal force from the runway upon the tires must be provided for, as well as frictional forces on the tires when the vehicle is in motion on the ground. Additional frictional forces result from braking during rollout. Runway unevenness and/or tire flex effects may be calculated for motion cues. At touchdown, high speed dynamic transients resulting from shock strut and tire flex motion at landing must be simulated.

At very low altitudes, at takeoff and landing, ground aerodynamic effects must be accounted for (air compression between orbiter and ground, etc.). Approximate magnitudes of ground effects at the landing angle of attack (13°) and speed

(300 ft/sec) are:

	50 ft Altitude	15 ft Altitude	10 ft Altitude
Change in lift Coefficient	+.02	+.15	±.25
Change in lift force	+7000 lb.	+50000 lb	+ 85,000 lb

(Altitudes those of wing aero center)

Thrust magnitudes from the boost SRM's, abort SRM's, Space Shuttle Main Engines, Orbital Maneuvering System Engines, and Air Breathing Engines must be resolved through the proper cant and/or gimbal angles, and the resultant forces found in the body axis system. Available canting and gimballing information for the above engines is as follows:

	CANT		GIMBAL		
	PITCH	YAW	TVC?	PITCH DEFL.	YAW DEFL.
MAIN ENGINES	UPPER 16.5° LOWER 10.5°	UPPER 0 LOWER INWARD 3.50	YES	±11°	±11°
OMS	11.5°	0	YES	±4°	0° to 12° Outward
BOOST SRM'S	NONE	11° Toward motor Centerline	NO	-	-
ABORT SRM'S	-3°	NONE	NO	-	-

Thrust from each RCS jet being fired must be applied as a force along the proper body axis, and included in the total body forces.

3.2.2.1.3.2 Rationale for Assumptions

Not applicable.

3.2.2.1.3.3 References:

1. 23 pp. 115 through 110, 59 pp. 2.4-15, 2.4-16
2. 166 pp. 2-58, 3-60
3. 166 pp. 3-75, 167, pp. 2-4
4. 166 pp. 2-34, 3-80
5. 166 pp. 3-81

REF. KEY

3.2.2.1.4 Payload Forces

3.2.2.1.4.1 Requirements

During operation of the payload manipulator system, significant forces may be exerted upon the shuttle orbiter, as well as the payload. Orbiter payload mass can reach slightly over
1 2000 slugs, while orbiter mass without payload will be about 5000 slugs. It is evident from conservation of linear momentum that the process of deploying such a payload could generate significant position and velocity perturbation upon the orbiter. Thus, forces on the orbiter body due to payload handling (and payload manipulator docking) operations must be included in the summing of total body forces acting upon the shuttle orbiter. Error or malfunction in payload handling operations could also result in low-speed collisions with detached payloads, which will result in forces(or momentum changes). Similarly, placement of a retrieved payload in the payload bay will also ordinarily result in small body force (momentum change) upon the orbiter at the time of contact.

3.2.2.1.4.2 Rationale for Assumptions

Not applicable.

3.2.2.1.4.7 References

1. 166 pp. 2-56

3.2.2.1.5 Docking Effects

3.2.2.1.5.1 Requirements

1 Maximum docking position misalignment of six inches and maximum relative velocity at contact of 0.5 $\frac{ft.}{sec.}$ are anticipated for orbiter vehicle design.

REF. KEY

2 The orbiter docking mechanism will be a neuter docking concept similar to that developed under the Advanced Missions Docking System Contract. The concept is illustrated below.

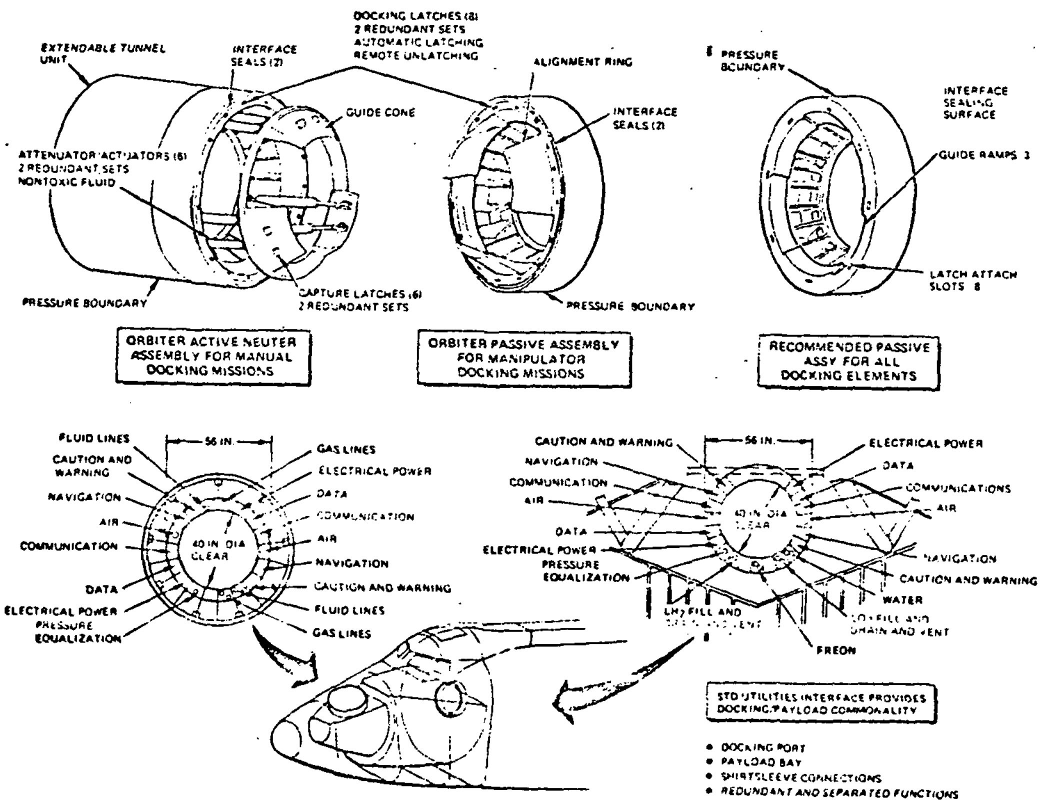

Figure 3-41. Neuter Docking Concept Meets All Requirements

The mechanism has redundant sets of attenuators with ten-inch stroke to meet misalignment requirements. A docking ring jettison charge holder, similar to that used on CSM, will be used for separation in emergency and contingency conditions.

Forces (momentum changes) upon the orbiter due to position misalignment and relative velocity at contact must be summed into the total orbiter body force, between contact and hard

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REF. KEY

dock. Attenuator effects must be included during docking and undocking.

3.2.2.1.5.2 Rationale for Assumptions

Not applicable

3.2.2.1.5.3 References

1. 20 pp. IV-12
2. 166 pp. 3-52, 3-53, 3-93

3.2.2.1.6 Staging Effects

3.2.2.1.6.1 Requirements

1 The SRM separation system will utilize two clusters of separation rockets to provide spacing between the SRM cases and the orbiter/external tank vehicle to satisfy abort ($t \geq 86$ sec.) and staging requirements. Each cluster of separation rockets contains three 27,000 lb. thrust motors. One cluster is located forward, one aft. The aft separation rockets are installed at a greater angle than the forward rockets, to account for residual thrust from the skewed main rocket nozzle. The separation rockets will fire for $.02 \pm .01$ seconds prior to SRM release, which is pyro actuated. Total
2 firing time is two seconds. Proximity aerodynamic effects are significant, especially during abort separation. The simulation should be able to identify recontact of SRM's
A with orbiter/external tank.

3 External tank separation is accomplished by a zero-force
4 release. An explosive-bolt system (triply redundant) is used for release of the abort SRM's.

REF. KEY

3.2.2.1.6.2 Rationale for Assumptions

A. The assumption is made that recontact dynamics need not be simulated. SRM recontact would appear to be a catastrophic situation. Under previous ground rules, catastrophic situations must be detected, but no further simulation requirements are made. SRM recontact would likely occur either on the orbiter wing or against the external tank. The consequences of collision damage to the orbiter wing are reasonably
5 obvious. The tank is of thin-wall aluminum monocoque construction, not designed for SRM impact. The consequences of a sudden large rupture of the LH_2 tank are also reasonably obvious.

3.2.2.1.6.3 References

1. 166 pp. 2-46, 2-47, 2-48, 3-83
2. 168 Sects. 9 and 10
3. 166 pp. 2-45, 2-46
4. 166 pp. 3-82, 3-83
5. 166 pp. 3-151, 3-152

3.2.2.1.7 Venting and Dumping

3.2.2.1.7.1 Requirements

The presence of hazardous fluids on-board and thermal constraints
1 on equipment require controlled venting of body cavities during boost and entry to meet limits on basic airframe pressure differential. Pressure differential control will be accomplished by a network of non-propulsive vent ports located on the leeward side of the vehicle.

REF. KEY

Residual and unused propellant in the external tank will be dumped and the tanks vented prior to tank separation. Dumping and venting will be accomplished through overboard
2 dump lines, and liquid propellant will also be dumped through the engines concurrently to settle remaining propellants. A velocity increment of up to 30 ft/sec may result from this thrust. LO_2 dumping will be accomplished first, followed by LH_2.

Following deorbit, downstream OMS propellant lines and engines are purged with residual helium. Engine valves are reclosed
3 prior to entry. Residual OMS propellants are dumped during entry following a normal mission. The system is sized to dump all OMS propellant in the pods prior to landing in a low altitude abort.

3.2.2.1.7.2 Rationale for Assumption

Not applicable.

3.2.2.1.7.3 References

1. 166 pp. 3-22
2. 166 pp. 2-45, 2-46, 3-61
3. 166 pp. 3-70

3.2.2.2 Trajectory Calculation Requirements

3.2.2.2.1 Orbiter

3.2.2.2.1.1 Requirements

The Equations of Motion simulation must maintain an accurate orbiter trajectory through all mission phases, reflecting the effects of all significant forces upon the orbiter. Given body forces, vehicle mass, and gravitational accelration, it must maintain orbiter inertial position velocity, and acceleration.

Before launch, orbiter inertial state must reflect earth rotational rate to a high degree of accuracy. During powered flight, relatively rapid change in mass and in direction (and, at times, magnitude) of body force requires high frequency of integration of forces and accelerations. During entry and atmospheric flight, possible rapid changes in the total body force vector also requires high integration rates. During orbital coast, resultant acceleration on the body is almost entirely that of gravity. Gravitational force consists chiefly of an inverse-square term, which is very amenable to exact closed-form analytic methods of propagation rather than open-loop integration. Gravitational force is also very stable, and can readily be approximated ahead. On the other hand, as discussed in Section 3.2.2.4, accuracy requirements during orbital coast are considerably more stringent. Thus, a lower frequency may be used for integration, but a scheme which maintains the predictable and slow-changing gravitational effects at very high precision is required. If a very low frequency is used, a simple high speed "interpolating" scheme may also be required to make state information available in the interim between the precise low-frequency calculations.

3.2.2.2.1.2 Rationale for Assumptions

Not applicable.

3.2.2.2.1.3 References

Not applicable.

3.2.2.2.2 Target Vehicles

3.2.2.2.2.1 Requirements

A number of situations may arise in which the inertial position and velocity of a vehicle other than the primary space shuttle vehicle must be maintained.

During the nominal launch and insertion profile, other separate vehicles are in the immediate vicinity of the shuttle vehicle at three points: Abort SRM jettison, separation from boost SRM's, and separation from the external tank. In each case, it would be desirable to verify that recontact does not occur, which would require maintenance for varying amounts of time of state vectors for these vehicles. In low altitude abort cases, it may be desirable to verify that recontact with the separated external tank/SRM assembly does not occur. In the cases of abort SRM's, boost SRM's, external tank/SRM assembly, or external tank jettison within the atmosphere, state would only be maintained for a short time (of the order of several seconds) to verify separation. The case of external tank jettison in orbit is somewhat different. If the deorbit mechanism fails, it may be necessary to maintain external tank state for a considerable period of time. Tank state will have to be maintained at least until after the next orbiter burn in this case, longer if that burn is small. In all of these launch phase cases, gravitational and aerodynamic forces must be considered, as well as residual main nozzle and separation rocket thrust in the case of the

boost SRM's and deorbit rocket thrust in the case of the external tank in orbit.

Translational state of payloads not attached to the vehicle must be maintained during the time they are being handled by the manipulator arms. Positions and velocities of deployed payloads should be maintained so long as visual or radio contact may be maintained. Accurate position and velocity of rendezvous target vehicles must be maintained when the target vehicle is within ranging distance (less than 300 n.mi.). Rendezvous target vehicle state should be maintained during any period at which the ground might uplink such information to the orbiter (i.e., starting some time before the beginning of the rendezvous sequence). Rendezvous target vehicles may include satellites to be retrieved or examined, other orbiters, the orbit-to-orbit-shuttle (OOS), and the space station. Extended periods of station-keeping may be required in some cases. In each of these cases, accurate gravitational forces must be included. Simplified aerodynamic forces will be adequate in most cases, but in the case in which the target vehicle is another orbiter in low earth orbit (possible on a rescue mission) or if extended station-keeping is required, target vehicle aerodynamics should be simulated as accurately as those of the primary orbiter. Payload manipulator force must be included when applicable. Target vehicle propulsion system forces must be included where applicable, (e.g., the OOS). It may also be necessary to simulate OOS burn targetting.

Upon such time as the TDRS (Tracking and Data Relay Satellite) system becomes operational for shuttle use, it will become necessary for the Equations of Motion system to keep track of the positions of each of the satellites in the system.

3.2.2.2.2.2 Rationale for Assumptions

Not applicable

3.2.2.2.2.3 References

1. 166, Page 3-99
2. 20, Page IV-26; 166 Pages 3-110, 3-111

3.2.2.3 Relative Translational States

3.2.2.3.1 Requirements

Relative translational states must be calculated during boost for any separated object being checked for recontact (e.g., Abort SRM's, boost SRM's, external tank, external tank/SRM assembly) until safe separation is verified. Relative translational states must be calculated for the external tank after nominal insertion from jettison to successful deorbit.

Relative state is required for payloads during and immediately after deployment. It is also required for rendezvous target vehicles during rendezvous final phase and during payload retrieval, docking, or station-keeping. Target vehicle relative state may also be found desirable during the period of rendezvous ranging (relative distance less than 300 n.mi.).

3.2.2.3.2 Rationale for Assumptions

Not applicable.

3.2.2.3.3 References

Not applicable.

3.2.2.4 Accuracy

3.2.2.4.1 Requirements

Precise accuracy requirements upon the Translational Equations of Motion derive largely from the need to be consistent with other systems which maintain vehicle state, most notably the guidance system. Requirements

arising from these needs are frequently more stringent than those deriving from such drivers as the need to present reasonable cues to the crew, the need to faithfully duplicate mission timelines, etc.

The Translational Equations of Motion/Guidance System interface is ordinarily through the IMU inertial accelerometers, which measure body accelerations in the current platform reference system. They obtain this acceleration from Equations of Motion, and output to the guidance system that value biased by appropriate accelerometer error sources. Thus, the gravitational calculations and velocity/position integrations are carried on in parallel. In the real world, after a period of time, the guidance state vector and the "true" state vector diverge slightly due to accelerometer error, approximations in gravitational calculation, numerical error in the guidance integration method, etc. The simulated Equations of Motion (EOM) should diverse in a similar manner from the guidance state vector. This can be accomplished only if the gravitational approximation in EOM is at least as accurate as that in guidance, and only if all error sources in the parallel portion of EOM are much smaller than the real world dynamics/guidance system divergence.

Insertion accuracy of the guidance system is expected to be:

	POSITION	VELOCITY
Downrange	.2 n.mi.	6 ft/sec
Crossrange	.2 n.mi.	10 ft/sec
Vertical	.2 n.mi.	10 ft/sec

(Velocities here also probably include tailoff dispersions, which involve unpredictable body accelerations. Thus, the divergence between simulated EOM and real-word dynamics (given the same body accelerations) should be considerably less than the above values (say one-fourth or less in position, less than that in velocity) during boost. The EOM velocity accuracy

requirement could be relaxed if good tailoff simulation is not required.

Probably the most critical orbital powered-flight accuracy requirement is on the deorbit burn. It is necessary, in a 500 x 500 n.mi. orbit, to limit downrange velocity error to less than 3 feet/sec. This value includes engine tail-off effect, so maximum permissible divergence between guidance and real-world state is substantially less. In the 500 x 500 case, deorbit delta V is 800 feet/sec, and, with one OMS engine out, burn time is 20 minutes. Moreover, the platform alignment with respect to which burn acceleration is measured was likely established sometime prior to the burn. Thus, it appears that something like the following is required: on board navigation accuracy within somewhat less than 1 foot/sec over an orbiter period of 30 minutes including 20 minutes of powered flight. If good engine tailoff simulation is required, the simulated EOM system should be well within this tolerance. If tailoff accuracy is considered unimportant, simulated EOM accuracy limits should still be no wider. During entry which is similar to powered flight in that substantial body accelerations are present, the guidance system must, at 100,000 feet, have the vehicle within 20 n.mi. and 130 feet/sec of the desired state. Again, navigation requirements will probably be more severe, though likely not as severe as for boost.

Reasonable requirements must also be placed on the accuracy of body accelerations fed to both EOM and the platform accelerometers during powered flight or entry. These accelerations are derived from forces fed to it from other systems (e.g., propulsion, aerodynamics, etc.), and from vehicle mass. As vehicle mass (powered flight) as well as the forces are probably changing rapidly, thus calculation must be done frequently

to avoid serious "sampling lag" error. The system is forgiving of small errors in body accelerations; the guidance system will erase them in actions imperceptible to crew members in powered-flight or entry. Also, fairly sizable dispersions exist in any case in several of the body forces. Problems will exist only if errors become sizeable enough to cause the guidance system, in erasing them, to move perceptibly outside the nominal flight envelope. Thus, if systems providing force values (propulsion, aerodynamics) have accuracy within their own nominal dispersions, mass is updated frequently, and body acceleration is calculated equally or more frequently; this should not be a serious problem during powered flight or entry.

Orbital coast presents different requirements. Boost lasts less than 10 minutes, orbital powered flight perhaps 20 minutes at worst, entry to 100,000 feet less than 30 minutes (navaid updates below that point should erase small discrepancies). Orbital coast(particularly if use of simulator"fast-time" is made)may last many hours. In so long a time small discrepancies may become very large ones. In orbit, not only must guidance/EOM discrepancies be simulated reasonably, but orbital decay and perturbation must be accurately portrayed. The results of even fairly small errors are likely to become guite obvious to the crew after a time. For example, in a 50 x 100 n.mi. orbit, it is necessary periodically to perform short orbital maintenance burns to counteract perturbing effects (especially drag). The necessary magnitude and frequency of these burns for a given attitude timeline is likely to be well known. If orbital coast integration and force simulation are not highly accurate, one of two things will happen:

1. If targetting equations operate directly or indirectly (via nav updates from sensors) using EOM data, burn delta V's will not match nominal values, or,
2. If nominal delta V's are burned, the orbit will not be maintained.

In either case, the discrepancies will be obvious to trainees in fairly short order. In a 50 x 100 orbit, a delta V of about 1.8 feet/sec at apogee will raise perigee by 1 n.mi. It seems desirable that, in such case, over one revolution, delta V discrepancies of no more than 0.1 foot/sec be caused by EOM. This would require position tolerance of 150 feet and velocity tolerance of 0.05 foot/sec. This tolerance would include all errors in gravity, aero drag, and the numerical integration scheme. It may be desirable in some cases to do considerably better than this. Error in relative state between two station-keeping vehicles over an orbit should be far smaller and should not change rapidly.

This requires that a station-keeping vehicle be simulated, in terms of aero, gravity, and integration, approximately as faithfully as the prime orbiter, unless station keeping continues only for a short time.

3.2.2.4.2 Rationale for Assumptions

Apollo and Skylab experience has indicated that considerable chagrin on the part of trainees and instructors results when simulated on-board, EOM derived, and nominal burn targettings do not agree quite closely. For close agreement, EOM, guidance, and nominal state vectors must be quite close. In Apollo and Skylab training, EOM derived targetting was used as a "reasonableness test" upon on-board targetting (or vice versa),just as,

during missions, ground RTCC targetting was used to check on-board targetting. Thus EOM was expected to be close enough to nominal to pass "reasonableness tests". The shuttle situation may be somewhat different. Due to high autonomy and parallel strings in on-board guidance, it is not clear whether RTCC targetting will have as much of a role to play. Thus, being mostly dependent on on-board guidance the crewmen may be more prone to apply as a "reasonableness test" the targettings they saw during training. This would require a high degree of fidelity in simulated EOM. A second related consideration is also significant. A simulation, like a true mission, must perform well in off-nominal conditions. Ordinarily, its off-nominal performance cannot be directly checked. It can only be inferred from performance under nominal conditions. However, since the simulation was verified for nominal conditions, instructors and trainees understandably assume it will perform better for them than in off-nominal conditions. Thus, they tend to expect much better than minimal performance in the nominal case, in hopes of ensuring at least minimal performance in off-nominal conditions. The requirements endeavor to keep these considerations in mind.

3.2.2.4.3 References

1. 166, Pages 2-11, 2-54, and 2-79
2. 166, Page 2-80.
3. 166, Pages 2-17, 2-77.

REF.
KEY

3.2.3 Rotational Equations of Motion

3.2.3.1 Moments

Moments arising from forces discussed in Section 3.2.2.1 are noted here, as well as moments arising from other sources. Discussions of the cause and nature of forces giving rise to moments are not repeated when covered in Section 3.2.2.1.

3.2.3.1.1 Dynamic Body Moments

3.2.3.1.1.1 Requirements

The Rotational equations of motion for a simulated shuttle vehicle must incorporate aerodynamic moments through much, perhaps all, of the anticipated flight envelope into attitude calculations. At very low altitudes, moments due to ground effects must be considered. Changes in pitching moment at landing angle of attack (13°) and speed (300 ft/sec) are:

	50 ft. alt.	-0.003
1	15 ft. alt.	-0.024
	10 ft. alt.	-0.038

(altitudes those of wing aero center)

Moments exerted by the ground upon the vehicle must be included. Moments arising from forces on the vehicle on the pad must be picked up (or moment calculation bypassed and inertial attitude and body rate updated regularly to account for earth rotation).

At horizontal takeoff, and during landing, moments resulting from normal and frictional runway forces must be accounted for. At touchdown, dynamics resulting from shock strut and tire flex effects must

REF.
KEY

be simulated. Rotational effects of runway unevenness and/or tire flex must be included.

Moments resulting from thrust of Space Shuttle Main Engines, Orbital Maneuvering System Engines, Reaction Control System Engines, boost and abort SRM's, and Air Breathing Engines must be included. In each case, the calculated forces must be applied at the proper point on the body, the orientation of that point with respect to the current c.g. position determined, and the resulting moment found. Positions of the aforementioned engines in the orbiter coordinate system are as follows:

Ref. Key			X (in)	Y (in)	Z (in)
	Main Engines	Lower	1468.3	± 52	335.8
		Upper	1445	0	433
2	ØMS Engines		1547	± 96	454
	Boost SRM's		1470	--	127
	Abort SRM's		1550	±168	352
	Air Breathing Engines		About 1300	_ 70.5	532.5

3 An RCS pod is located in each ØMS pod. A third RCS pod is located in the vehicle nose.

3.2.3.1.1.2 Rationale for Assumptions

Not applicable.

3.2.3.1.1.3 References

1. 23 P. 117
2. 167 PP. 2-4, 2-7; 168 PP. 2-4, 2-6, 2-7
3. 166 P. 3-63

REF. KEY

3.2.3.1.2 Payload Moments

3.2.3.1.2.1 Requirements

The payload manipulator system will, while in operation, exert significant moments upon the shuttle orbiter. Forces due to low-speed collisions with payloads, and forces resulting from small relative velocities at payload placement will also ordinarily possess moment arms, and may be significant. Any small relative angular rates at the time of placement should also be accounted for. Moments due to motion in attached payloads should be simulated (e.g., telescope pointing on an astronomy sortie mission).

3.2.3.1.2.2 Rationale for Assumptions

Not applicable.

3.2.3.1.2.3 References

Not applicable.

3.2.3.1.3 Docking Effects

3.2.3.1.3.1 Requirements

1 Since the docking mechanism is located far from the vehicle c.g., forces due to relative velocity and position misalignment at contact will ordinarily have significant moment arms. Angular misalignments of up
2 to five degrees and roll misalignments of up to seven degrees may occur at docking. Angular velocities of 1 deg/sec for the active vehicle, and .1 deg/sec for the passive vehicle are other maximum values. The dynamics of the guides and attenuators of the docking mechanism (see Section 3.2.2.1.5) in nulling these errors must be properly simulated. Moments arising from the use of the contingency explosive charge docking ring release should be simulated.

REF. KEY

3.2.3.1.3.2 Rationale for Assumptions

Not applicable.

3.2.3.1.3.3 Data References

1. 167 P. 2-17; 164

2. 20 P. IV-12

3.2.3.1.4 Staging Effects

3.2.3.1.4.1 Requirements

Moments arising from all staging forces, including separation rockets and the pyro actuated release mechanisms, must be considered. Proximity aerodynamic moments are significant, particularly during abort separation. The simulation must maintain sufficient attitude information to identify recontact of SRM's with orbiter/external tank.

3.2.3.1.4.2 Rationale for Assumptions

Not applicable.

3.2.3.1.4.3 References

Not applicable.

3.2.3.1.5 Venting and Dumping

3.2.3.1.5.1 Requirements

Moments arising from body cavity venting forces during boost and entry may be significant. Moments arising from dumping and venting of the external tank after insertion must be simulated. Following deorbit, moments arising from purging of ØMS propellant lines and engines must be considered. Dumping of residual ØMS propellant during entry or after low altitude aborts may create significant moments.

REF.
KEY

3.2.3.1.5.2 Rationale for Assumptions

Not applicable.

3.2.3.1.5.3 References

Not applicable.

3.2.3.2 Rotational State Calculation Requirements

3.2.3.2.1 Orbiter

3.2.3.2.1.1 Requirements

The Equations of Motion simulation must, given all moments on the vehicle, and moments and products of inertia, accurately determine body angular acceleration, and maintain current values of body rate and inertial attitude.

Before launch, orbiter rotational state must reflect the effects of earth rotational rate to a high degree of accuracy. During powered flight, relatively rapid change in mass properties and moment vectors require high frequency integrations to maintain rotational state. During entry and atmospheric flight, possible rapid change in aerodynamic moments also require high integration rates. Moments will be small during orbital coast, but may still change rapidly. Rates will normally be very small, but could become considerable in malfunction cases. Thus, high integration rates are probably required for rotational state during orbit as well. The further requirement that the equations of motion respond properly to very small moments is also added.

At the minimum, the equations of motion must make available to other systems the orbiter body rates, and the transformation matrix between the inertial reference coordinate system and the body coordinate system. Other related tasks may be performed in Equations of Motion, or in another system.

3.2.3.2.1.2 Rationale for Assumptions

Not applicable

3.2.3.2.1.3 References

Not applicable.

3.2.3.2.2 Target Vehicles

3.2.3.2.2.1 Requirements

Attitude and angular rate information is required, at times, for several vehicles other than the orbiter. To some extent, such information is needed for nearly all of the vehicles cited in Section 3.2.2.2.2.

To obtain reasonable aerodynamic forces and moments, the attitudes of all vehicles operating within the atmosphere must be maintained. Moreover, attitude must be maintained in the case of any vehicle which may recontact the prime space shuttle vehicle, in order to check for recontact. Thus, if recontact simulations are performed for the following vehicles cited in Section 3.2.2.2.2, attitudes must also be maintained: Abort SRM's, boost SRM's, SRM/external tank assembly, and external tank jettison within the atmosphere.

Vehicles which may undergo powered flight must have attitude known. This would include the external tank in orbit (to properly simulate deorbit burn), and the ØØS. In most cases, attitudes of vehicles possessing attitude control systems will have to be simulated, as well as the attitude control systems themselves (perhaps in simplified terms). And, in any case, when close visual contact exists, rates and attitudes of vehicles must be simulated. Thus, rotational states of payloads not attached to the prime orbiter must be maintained while visual contact exists. Similarly, rotational states of payloads to be

REF. KEY

retrieved, and target vehicles to be docked must be available while in proximity. If extended station-keeping is a mission feature, the simulation must be of a reasonably high order of accuracy.

3.2.3.2.2.2 Rationale for Assumptions

Not applicable.

3.2.3.2.2.3 References

Not applicable.

3.2.3.3 Relative Rotational States

3.2.3.3.1 Requirements

Relative rotational state should be available during boost for any separated object being checked for recontact (e.g., Abort SRM's, boost SRM's, external tank, external tank/SRM assembly) until safe separation is verified. Relative rotational state for the external tank in orbit from jettison to deorbit may also be necessary.

Relative rotational state for payloads during and after deployment until visual contact is lost should be available, as well as for rendezvous target vehicles during rendezvous final phase and during retrieval, docking, or station-keeping.

3.2.3.3.2 Rationale for Assumptions

Not applicable.

3.2.3.3.3 References

Not applicable.

3.2.3.4 Accuracy

3.2.3.4.1 Requirements

Two factors comprise the primary drivers for accuracy requirements on Rotational Equations of Motion:

REF.
KEY

1) To adequately simulate effects of IMU alignment error, bias, drift, etc.; numerical errors within the sections of Rotational Equations of Motion parallel to the IMU simulation must be much smaller.

2) Simulated attitude control system and vehicle response must be reasonably similar to real world response.

During powered flight and entry, the vehicle trajectory is largely a function of body forces. Before their effects can be utilized by Translational Equations of Motion, the forces must be transformed into its reference system. This is accomplished using body attitude (relationship between the body-fixed coordinate system(s) and the reference system) maintained by Rotational Equations of Motion. However, during these periods, the control system will rapidly drive out any small errors it senses. Thus, if the attitude known to the control system and the EØM attitude are close enough (they will be if requirement (1.) is satisfied), and if errors are not so large as to cause perceptible anomalies in control response (they will not if requirement (2.) is met), this requirement should cause no serious difficulty.

IMU alignment on the pad will be accurate to within 75 arc-
1 seconds (.021 deg) in azimuth and 30 arc-seconds (.0083 deg) in level. Other G&N system accuracies and errors are tabulated below:

REF. KEY

Component	Characteristic	Equipment Capability (1σ)	Error Budget (1σ)
IMU			
Accelerometer	Bias (μg)	25	100
	Scale factor (%)	0.0025	0.0150
	Input/Output axis misalign (sec)	10	40
Gyro	Scale factor (%)	0.04	0.10
	G-insens drift (°/hr)	0.03	0.05
	Mass unbalance input axis (°/hr/g)	0.1	0.1
	Mass unbalance spin axis (°/hr/g)	0.03	0.10
	Anisoelastic (°/hr/g^2)	0.003	0.100
Gimbal	Readout (two-speed) (sec)	25	72
Horizon sensor	Angle (min) (limited by horizon definition)	6	9
Star sensor	On-Axis random and bias (min)	0.5	0.6
Backup optical sight	Instrument, alignment, limit cycle (min)	11	12
IMU/horizon sensor/star sensor	Alignment (min)	1.0	1.2
Backup sensor			
Gyro	Scale factor (%)	0.05	0.065
	Bias (°/hr)	0.25	0.50
Accelerometer	Scale factor (%)	0.05	0.10
	Bias (μg)	400	500

2

Rotational EØM and the simulated IMU will probably interface through body rates, which the IMU will obtain from Rotational EØM. Thus, numerical errors in Rotational EØM attitude calculations must be much smaller than the above G&N errors in order to correctly simulate IMU dispersions.

The extent of accuracy required to simulate control response reasonably will be a function of anticipated simulator use, whether as a crew training simulator, or as both an engineering and training simulator. For example, in one case, the simulator would only be required to exhibit during orbital coast (for a given deadband) RCS firings which would not be perceptibly different to the crew from real world firings. In the other case, RCS fuel consumption studies might be required, placing much stiffer requirements on accuracy. Possible error sources which could affect response (and still leave requirement 1.) satisfied) include:

REF.
KEY

- inaccurate body forces and aero moments
- inaccurate or lagged c.g. location
- inaccurate or lagged inertial properties
- numerical lag in Euler equations due to use of past values of rates
- integration of angular accelerations to obtain rates

If propulsion forces, aero moments, etc., supplied by other subsystems are within their nominal dispersions, they should cause no serious problems. During periods of rapid mass property change (e.g. boost), the c.g. and inertial properties must be updated frequently. Numerical and sampling error in the Euler equations and integration must be maintained at a low level of inaccuracy.

3.2.3.4.2 Rationale for Assumptions

Not applicable.

3.2.3.4.3 References

1. 166 P. 2-79
2. 166 P. 3-99

REF. KEY

3.3 Mass Properties

3.3.1 Orbiter

3.3.1.1 Requirements

The orbiter mass properties may change substantially during a mission. Portions of the vehicle showing mass changes include:

- Payload Bay (payload deployment and retrieval)
- OMS Tanks (propellant use and dumping)
- RCS Tanks (propellant use and dumping)
- Cryogenic Tanks (usage of contents)
- Crew Cabin (personnel off-loading and on-loading)
- MPS Lines (propellant and dumping)
- APU Tanks (usage of contents
- Water System
- ECS GN2 Tanks
- Helium Pressurization System

Vehicle configuration variables include:

- Payload Doors Open/Close
- Landing Gear Up/Down
- Payload Manipulator

Each of these will influence orbiter mass properties to some extent. For example, deploying and using the manipulator arm
1 will affect the vehicle c.g., and moments and products of inertia. Contents (and for the OMS, number) of tanks may vary from mission to mission.

REF. KEY

The orbiter mass properties simulation must be sufficiently flexible to allow mission-to-mission change in mass properties due to differing payloads and tankage characteristics. It must also properly simulate mass properties changes due to such mass and configuration changes as those cited above, for any full mission.

Orbiter mass properties data is available for certain mission modes in the 40,000 lb. payload mission.

REF. KEY	SUMMARY	WEIGHT
	Wing Group	10,306
	Tail Group	2,440
	Body Group	35,348
	Induced Envir. Protect.	24,918
	Landing, Docking	11,230
	Propulsion, Ascent	24,588
2	Propulsion, Cruise	62
	Propulsion, Auxiliary	6,426
	Prime Power	3,467
	Electrical Conv. & Dist.	3,970
	Hydraulic Conv. & Dist.	2,520
	Surface Controls	1,849
	Avionics	4,157
	Envir. Control	3,705
	Personnel Provision	1,603
	Growth	11,946
	Total Dry Weight	148,535
	Personnel	1,211
	Cargo	40,000
	Residual Fluids	2,649
	Total Inert Weight	192,395
	Reserve Fluids	773
	Inflight Losses	3,409
	Propellant Used-Maneuv/ACS	17,313
	Total Lift-Off Weight	213,890

Summary of Mass Properties (40k polar mission)

	Weight (lb)	Center of Gravity (Inches) X	Y	Z	Moment of Inertia Slug ft^2 X 10^6 IXX	IYY	IZZ	Product of Inertia Slug ft^2 X 10^6 IXY	IXZ	IYZ
	(Payload cg 30 ft. from payload bay forward wall)									
Lift-off	213,890	1082.1	0.2	381.2	.821	7.130	7.576	-.007	.207	neg 'l
* Post-Tank Jettison	213,585	2101.2	0.0	694.3	.821	7.126	7.373	---	---	---
Entry	195,583	1070.2	0.2	380.0	.790	6.224	6.448	-.007	.157	neg 'l
Landing	193,168	1068.2	0.2	377.5	.803	6.152	6.359	-.007	.142	neg 'l
Entry-No P/L	155,583	1103.6	0.3	374.9	.749	5.646	5.875	-.007	.185	neg 'l
Landing-No P/L	153,168	1101.7	0.3	371.6	.760	5.579	5.790	-.007	.173	neg 'l
	(Payload cg 46 ft. from payload bay forward wall)									
Landing	193,168	1108.0	0.2	377.5	.803	5.980	6.186	-.007	.179	neg 'l

* C.G.'s in integrated vehicle coordinate system; others are orbiter coordinate system.

3.3.1.2 Rationale for Assumptions

Not applicable.

3.3.1.3 References:

1. 166 pg. 3-68
2. 166 pg. 2-17; 167 pg. 2-16, 2-17

REF. KEY

3.3.2 Solid Rocket Motors

3.3.2.1 Requirements

The shuttle vehicle, at launch, possesses two abort SRM's and two boost SRM's. Each is jettisoned during boost. Mass property simulations of both are required until shortly after jettison. Each, once ignited, has fairly rapidly changing mass properties, as propellant is depleted. The abort SRM's burn at near full thrust for about 20 seconds. Sample mass properties for both abort SRM's are:

1

	Weight (lb)	Center of Gravity (Inches) X	Y	Z	Moment of Inertia Slug $ft^2 \times 10^6$ IXX	IYY	IZZ	Product of Inertia Slug $ft^2 \times 10^6$ IXX	IXZ	IYZ
ASRM at Lift-Off	92,700	2437.9	0.0	663.6	.797	.103	.888	.0	neg'l	.0
ASRM 1/2 Prop. Burned	56,200	2441.5	0.0	663.8	.477	.067	.537	.0	neg'l	.0
ASRM Prop. Burned	19,700	2458.5	0.0	664.8	.158	.030	.185	.0	neg'l	.0

2

The boost SRM's burn for 112 seconds - near full thrust for 102 seconds.

1

Sample of mass properties for both are:

	Weight (lb)	Center of Gravity (Inches X	Y	Z	Moment of Inertia Slug $ft^2 \times 10^6$ IXX	IYY	IZZ	Product of Inertia Slug $ft^2 \times 10^6$ IXY	IXZ	+YZ
Lift-Off	2,563,700	1889.8	0.0	448.1	33.697	72.175	104.017	0.0	.015	0.0
1/2 Prop. Burned	1,449,700	1889.2	0.0	448.2	19.115	44.328	62.320	0.0	.015	0.0
Prop. Burned	335,700	1970.9	0.0	448.9	4.534	16.059	20.201	0.0	.010	0.0

2

REF. KEY

3.3.2.2 Rationale for Assumptions

Not applicable

3.3.2.3 References

1. 166 pg. 3-83
2. 168 pg. 2-18

3.3.3 External Tank

3.3.3.1 Requirements

The shuttle vehicle obtains boost propellant and oxidizer for its main propulsion system from an external tank. External tank mass properties must be known so long as the tank is attached to the orbiter, and so long as its state vector must be maintained after jettison. Propellant flow effects during boost, and effects of dumping of residual propellant upon tank mass properties must be simulated.

3.3.3.2 Rationale for Assumptions

Not applicable.

3.3.3.3 References:

Not applicable.

3.3.4 Payload

3.3.4.1 Requirements

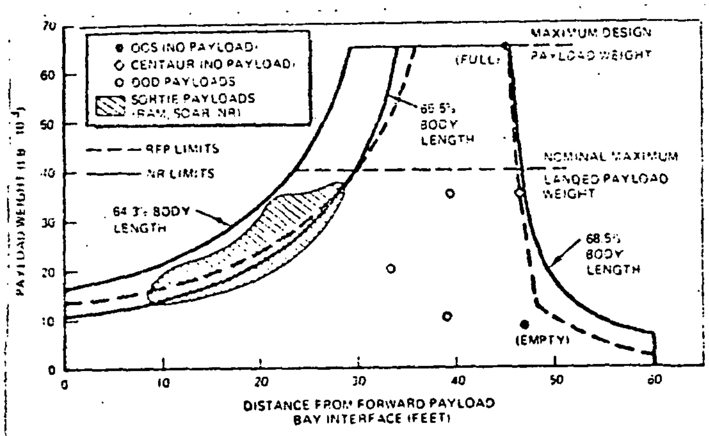
Mass properties of attached payloads must be available in the calculation of total vehicle mass properties (mass, c.g., moments and products of inertia). Provision must be made to deduct mass properties of payloads, when

REF.
KEY

released, from total vehicle mass properties, and to include mass properties of retrieved payloads when attached. Masses of several payloads are included in Section 3.1.1.2. Any significant changes in payload mass of attached payloads during the mission should be simulated (e.g., crew ingress/egress of manned payload). Change in configuration of attached payloads during mission may be significant (e.g., telescope pointing on astronomical sortie mission).

Permissible payload c.g. envelope (with examples of possible payloads) is:

1



3.3.4.2 Rationale for Assumptions

Not applicable

3.3.4.3 References:

1. 166 pp. 3-155

KEY
REF.

3.3.5 ABPS

3.3.5.1 Requirements

When carried, mass properties due to the presence of the ABPS must be simulated. Two ABPS engines will be carried
1 on some orbital flights. The ABPS will assume two configurations. During most of the flight (boost through entry), the ABPS will be stored in the aft payload bay. During approach, they will be deployed on the upper fuselage. Simulated mass properties of the system should reflect the appropriate configuration. Changing mass properties of the system during the deployment process may also have to be simulated. Changing mass properties of the ABPS tanks during engine operation must be simulated.

The total dry weight of the orbital ABPS is 12,385 lbs. fuel weight carried will ordinarily be 13,681 lbs., although tankage
2 will be sized to contain 22,565 lbs. Full tank capacity may be used for early development flights.

3.3.5.2 Rationale for Assumptions

Not applicable

3.3.5.3 References:

1. 166 pp. 3-75, 3-76
2. 166 pp. 3-77, 3-78

3.3.6 Ferry ABES

KEY REF.

3.3.6.1 Requirements

During horizontal flight, mass properties of the four-engine ferry ABPS must be included in total mass properties. Changing mass properties of the ABPS tanks during engine operations must be simulated.

Total dry weight of the ferry ABPS is 22,529 lbs. Fuel weight normally carried will be 52,000 lbs., but tank capacity for 67,470 lbs

1 will be provided. Full capacity may be used during flight testing.

3.3.6.2 Rationale for Assumptions

Not applicable.

3.3.6.3 References:

1. 166 pg. 3-77

3.3.7 Total Vehicles

3.3.7.1 Operational Space Vehicle

3.3.7.1.1 Requirements

The space shuttle operational vehicle includes the following configurations on a space mission:

- orbiter + tank + boost SRM's + Abort SRM's
- orbiter + tank + boost SRM's
- orbiter + tank
- orbiter + Abort SRM's (abort only)
- orbiter only

The orbiter may or may not include the orbital ABPS. At a given point it may have one or more payloads, or no payload.

KEY REF.

Given the current configuration, and the mass properties of the items discussed in Section 3.5.1 through 3.5.6, the current values of vehicle mass, vehicle center of gravity, moments of inertia, and products of inertia must be available to rotational and translational equations of
1 motion. The values must be calculated sufficiently frequently and/or smoothed in order to avoid "sampling lag" in equations of motion.

Several sample values of mass properties for several points and configurations during a 40,000 lb. payload polar mission are available.

		Center of Gravity			Moment of Inertia			Product of Inertia		
	Weight (lbs.)	(Inches) X	Y	Z	Slug $ft^2 \times 10^6$ Ixx	Iyy	Izz	Slug $ft^2 \times 10^6$ Ixy	Ixz	Iyz
(Orbiter + Tank + Boost SRM's + ASRM's)										
Lift-Off	4,656,739	1558.9	0.4	446.0	40,292	373,796	402,064	.085	21,330	.034
ASRM Jettison	3,738,024	1511.9	0.5	446.7	29,454	314,760	332,819	.104	20,381	.034
(Orbiter + Tank + Boost SRM's)										
ASRM Jettison	3,645,324	1488.3	0.5	441.2	27,693	296,100	314,337	.113	16,264	.036
Max Q	3,137,564	1453.9	0.6	441.3	22,000	258,924	271,815	.126	15,480	.036
Max Accel.	2,123,554	1329.5	0.8	441.3	10,553	178,137	180,276	.174	14,017	.036
Staging	1,994,125	1309.8	0.9	441.5	9.220	166,529	167,424	.181	13,796	.036
(Orbiter + Tank)										
Staging	1,658,425	1176.0	1.1	440.1	4,682	112,386	109,144	.233	13,363	.037
Tank 2/3 Full	1,436,700	1231.5	1.3	446.2	4,583	103,861	100,717	.209	12,601	.036
Tank 1/3 Full	873,367	1418.1	2.2	476.0	4,155	72,257	69,540	.133	9.928	.024
Burnout	310,034	1933.6	5.7	613.9	2,181	20,107	19,372	-.069	2.508	-.031
(Orbiter + Abort SRM's)										
Lift-Off	307,610	2204.0	0.0	685.0	1,632	8,843	9,861	-.008	.063	neg'l
ASRM 1/2 Burned	271,110	2173.3	0.0	687.9	1,308	8,337	9,045	-.008	.107	neg'l
ASRM Burnout	234,610	2133.0	0.0	691.8	0.983	7,686	8,083	-.008	.165	neg'l

3.3.7.1.2 Rationale for Assumptions

Not applicable

3.3.7.1.3 References:

1. 168 pg. 2-17, 2-19

KEY REF

3.3.7.2 Operational Ferry

3.3.7.2.1 Requirements

Operational ferry configuration configuration for the orbiter will have OMS pods removed, and the ferry ABES system installed
1 rather than the orbital ABES. Total weights during a nominal ferry mission (no loiter) are: (1st figure at beginning of phase, last at end)

	Ground	216,500 - 210,500 lbs.
2	Climb	210,500 - 202,000 lbs.
	Cruise	202,000 - 171,800 lbs.
	Descent	171,800 - 171,500 lbs.
	Approach & Land	171,500 - 171,300 lbs.

An additional fuel budget of 6800 lbs. is provided for up to 20 minutes loiter time.

The simulated mass properties system must be able to provide vehicle mass, center of gravity, and moments and products of inertia for the operational ferry configuration of the Space Shuttle Vehicle.

3.3.7.2.2 Rationale for Assumptions

Not applicable.

3.3.7.2.3 References:

1. 167 pg. 4-12
2. 167 pg. 4-17

KEY REF.

3.3.7.3 HFTS

3.3.7.3.1 Requirements

The orbiter used for horizontal flight testing will contain only those subsystems required for horizontal flight, as well as certain developmental subsystems which will be deleted from the operational vehicle. The configuration is summarized below:

1

SUBSYSTEM CONFIGURATION

ORBITER NO. 1			
TPS	• APPROX 10% LEADING EDGES, FWD AND UNDERSIDE AREAS, FAIRED CONTOURS	PWR DISTR AND CONTROLS	• DELETE INVERTERS, SEQUENCERS AND BATT CHARGER
PAYLOAD HANDLING	• DELETE DEPLOY/HANDLING PROVISIONS	HYDRAULIC	• DELETE MPS, GIMBAL AND ABPS DEPLOY ACTUATORS
GN&C	• AERO-GUIDANCE, NAVIGATION AND CONTROL PROVISIONS	CREW PROVISIONS	• ADD EJECTION SEATS (2)
COMM AND DATA	• INSTALL AIRCRAFT ANTENNAS IN PLACE OF OPNL EQUIP • DELETE SPACE GROUND LINK SUBSYS • ADD DFI	ECLSS	• DELETE FOOD, WATER, WASTE MANAGEMENT, SUBLIMATOR, RADIATORS, HYDRAULIC AND FUEL CELL HEAT EXCHANGERS
POWER GENERATION	• DELETE FUEL CELLS AND APU'S • ADD ENGINE DRIVEN HYD PUMPS AND GENERATORS	DISPLAYS AND CONTROLS	• DELETE MANIPULATOR STA D&C, PAYLOAD MONITORING, AND SPACE OPERATIONS D&C
ABPS	• FERRY KIT	OMS/RCS	• PODS REMOVED • MASS BALANCED, FAIRINGS
MPS	• MASS SIMULATED		
ORBITER NO. 2			
OPERATIONAL EXCEPT FOR 1. ADD SUBSYSTEM SEQUENCE CONTROLLER 2. ADD EJECTION SEATS (2) 3. INSTALL ABPS FERRY KIT 4. ADD DEVELOPMENT FLIGHT INSTRUMENTATION 5. OMS/RCS PODS REPLACED BY MASS BALANCE/FAIRINGS			

The simulated mass properties must be able to properly simulate the mass properties of the HFT configured orbiter.

3.3.7.3.2 Rationale for Assumptions

Not applicable

3.3.7.3.3 References:

1. 166 pp. 1-12; 178 pg. 7-5

KEY REF

3.3.7.4 VFT

3.3.7.4.1 Requirements

Configuration for vertical flight tests will be the same as that of the operational vehicle except for the inclusion of:

1. Development flight instrumentation
2. Ejection seats
3. Additional communication equipment

Configuration is described below:

Subsystem	Orbiter Configuration		
	Flight Test		Operational Suit.
	UVF	Flt S1-S6	Flt S7-S11
Structure	Operational	Operational	Operational
Landing	Operational	Operational	Operational
TPS	Operational	Operational	Operational
Payload accomm	Operational	Operational	Operational
MPS	Operational	Operational	Operational
RCS	Operational	Operational	Operational
OMS	Operational	Operational ΔV kits flts S4 and S5	ΔV kit on flts S7, S8, S9 and S11 Operational
ABPS	Operational	Operational on flts S1-S5 Remove on flt S6	Removed
GN&C	Add subsystem sequence controller	Operational	Operational
Comm and tracking	Add updata link S-band transmitter TV camera and viewer	Same as UVF	Same as UVF
D&C	Operational	Operational	Operational
Instrum	Add DFI	Same as UVF	Same as UVF
ECLSS	Operational	Operational	Operational
Power gen	Operational	Operational	Operational
EPD&C	Operational	Operational	Operational
Hydraulics	Operational	Operational	Operational
Crew provisions	Add eject seats	Same as UVF	Operational

The simulated mass properties system must be able to properly simulate the mass properties of the VFT configured orbiter.

3.3.7.4.2 Rationale for Assumptions

Not applicable

KEY REF.

3.3.7.4.3 References:

1. 166 pp. 1-12; 178 pp. 7-9

3.3.8 Retrieval Satellites

3.3.8.1 Requirements

Mass properties of retrieval satellites must be simulated during the time when their equations of motion are maintained, if body forces (propulsion, aero, etc.) or moments are significant. Provision must be made to adequately simulate mass property change of satellites whose properties are not constant (e.g., the ØØS during a burn). Masses of selected retrieval satellites are listed in Section 3.1.1.2.

3.3.8.2 Rationale for Assumptions

Not applicable

3.3.8.3 Data References:

Not applicable.

3.3.9 Deployed Satellites

3.3.9.1 Requirements

Mass properties of deployed satellites must be simulated during the time when their equations of motion are maintained if any significant body forces or moments (propulsion, aero, etc.) can exist. Provision must be made to adequately simulate mass property change of satellites whose properties are not constant. Masses of selected deployed satellites are listed in Section 3.1.1.2.

3.3.9.2 Rationale for Assumptions

Not applicable

3.3.9.3 Reference:

Not applicable

KEY
REF.

3.3.10 Space Station

3.3.10.1 Requirements

Mass properties of the space station must be maintained during the time when its equations of motion are simulated. Provision must be made to adequately simulate mass property change (e.g., module attach/separate, jet firings).

Some mass information is included in Section 3.1.1.2.

3.3.10.2 Rationale for Assumptions

Not applicable

3.3.10.3 References:

Not applicable

REF. KEY

3.4 Aerodynamic Characteristics

3.4.1 General Requirements

167 The aerodynamic forces and moments generated by a vehicle depend upon vehicle geometry, altitude, and velocity and attitude relative to the atmosphere.

For analysis the aerodynamic forces and moments are referenced to the vehicle body axes or to the stability axes based on the relative wind direction. Data is usually based on wind tunnel studies over a large range of Mach numbers and relative wind directions (angle of attack and sideslip angle). Data are presented in the form of dimensionless aerodynamic coefficients, defined as follows:

C_L Lift Coefficient (stability axes)
C_D Drag Coefficient (stability axes)
C_N Normal Force Coefficient (body axes)
C_A Axial Force Coefficient (body axes)
C_Y Side Force Coefficient
C_l Roll Moment Coefficient
C_m Pitch Moment Coefficient
C_n Yaw Moment Coefficient

The choice of using body or stability axes is arbitrary, usually dependent upon what is being analyzed.

In order to determine the generated forces and moments several aerodynamic parameters must be calculated. These include:

REF. KEY

167 α - angle of attack: the angle between the relative wind and a longitudinal reference line.

β - sideslip angle: the angle between the relative wind and the vehicle vertical plane of symmetry.

M - mach number: the ratio of the speed of the relative wind to the local speed of sound.

$\bar{q}$ - dynamic pressure: the pressure generated by the kinetic energy of the impacting airstream.

These parameters allow calculation of the forces and moments generated at any particular instant.

Forces are calculated by the equation:

$$F = \bar{q}\ S\ C_F$$

where: S is a reference area;

and moments, relative to some reference point, are calculated by the equation:

$$M = \bar{q}\ S\ l\ C_M$$

where: l is a reference length.

In order to determine the vehicle Mach number and dynamic pressure, a mathematical model of the atmosphere is required. There are several existing Standard Atmospheres which give tables of density, speed of sound, and other parameters as functions of altitude. Some Standard Atmospheres include variations due to seasonal changes and the exact shape of the earth, but these effects are usually ignored.

REF. KEY

167 In addition to the Standard Atmosphere model, perturbations due to winds, temperature and pressure variation, and the earth's rotation are present. The effects of winds and weather (temperatures and pressure changes) are significant at low altitudes, but their importance diminishes with altitude.

3.4.2 Aerodynamic Flight Regimes

167 During the shuttle mission profile three principal aerodynamic modes are encountered: (1) Launch and abort; (2) Orbital flight; and (3) Orbiter only. Each of these modes is unique; the forces and moments generated and the factors that produce them vary.

3.4.2.1 Launch and Abort

168 In this mode the vehicle flies almost directly into the relative wind (low angle of attack and sideslip) at Mach numbers ranging from zero to orbital velocity.

The vehicle is aerodynamically stable in pitch and yaw in this mode, until just prior to orbit insertion. This will not be a problem because the aerodynamic forces and moments generated are not significant. The vehicle exhibits neutral stability in roll in this mode, but this is of little consequence, since even large roll excursions are not detrimental to the ascent trajectory.

Data for this mode is usually generated in the body axes frame of reference. Axial force or drag is the predominant force, and the moments are generally small, due to the low angles of attack and sideslip. Dynamic derivatives are usually insignificant due to the low attitude rates. The shift in center of gravity as the vehicle's fuel is depleted affects the aerodynamic moments, but

REF. KEY

168 since the coefficients are usually based on fixed aerodynamic center this is not considered a problem. The problem that does occur in the launch and abort mode is the changes in vehicle configuration as the ASRM's, SRM's and the external tank are jettisoned. This produces significant changes in the vehicle aerodynamic configuration. In addition the changes in aerodynamic coefficients during separation is significant. These forces and moments due to proximity are usually estimated by adding an incremental coefficient due to the proximity of the jettisoned component. There are four configurations to be considered during the launch and abort mode: (1) The total vehicle; (2) ASRM staged (orbiter + SRM + Tank); (3) SRM Staged (Orbiter + tank); and (4) Orbiter only. There are normal mission times at which these configurations are assumed, due to the schedule for jettisoning components. In the event of an abort, dependent upon mission time, and possibly the pilot's judgement, the configurations may not follow the usual sequence.

3.4.2.2 Orbital Flight

59 In this mode the vehicle may assume any attitude with respect to the relative wind. Mach number is approximately fixed, so aerodynamic coefficients vary only with angle of attack and sideslip angle. The primary aerodynamic force is drag, resulting in eventual orbital decay.

The vehicle is essentially neutrally stable during orbital flight. Aerodynamic forces and moments generate small perturbations that must be cancelled by firing RCS jets if attitude is to be held fixed.

REF. KEY

59 Although all forces and moments in this regime are small, they will have noticeable effects in proximity to, or when docked with, payloads, due to the effects of even small perturbations on orbital mechanics.

Data for this regime are usually based on the capture area of the vehicle. Thus the exact shape and dimensions of the vehicle, or payloads, is less important than the impact area. A majority of the payloads considered for the shuttle are cylindrical; airload data for any payload can be scaled for any other payload without excessive error. The case of docking with another shuttle vehicle or with a large space station which may mask the vehicle from the relative wind needs to be considered, since aerodynamic forces are generated by the combined vehicle capture area.

The aerodynamic effects on the target vehicle must also be known in order to provide estimation of rendezvous and docking effects.

3.4.2.3 Orbiter Only

167 Orbiter-only flight is concerned with the flight characteristics of the orbiter from the beginning of re-entry (at about 400,000 feet altitude) to landing. It also includes any subsonic flight of the shuttle vehicle.

This mode is characterized by having the bulk of the aerodynamic effects generated by the basic orbiter, with incremental effects caused by control deflections, rotational rates, and configuration changes. The data for this regime are extensive, usually tabulated as functions of Mach number, angle of attack, sideslip angle, control deflections, and ground promixity.

REF. KEY

167 Mach numbers range from zero to about 25, while angle of attack generally varies from approximately zero to 60 degrees. The orbiter exhibits three areas of aerodynamic instability:

1. Longitudinal instability at high Mach/low alpha
2. Longitudinal instability at low Mach/high alpha
3. Directional instability, characterized by severe Dutch Roll, at hypersonic and high supersonic speeds

The longitudinal instability areas are avoided when using a nominal entry trajectory. The directional instability is overcome by the use of RCS jets for yaw control during re-entry. Data may be generated using either body or stability axes.

There are five configurations to be considered for the Orbiter-only mode: (1) Re-entry; (1) Approach and Landing; (3) Ferry Configuration; (4) Horizontal Flight Test Configuration; and (5) Vertical Flight Test Configuration. It is expected that the aerodynamic characteristics of these various configurations will be close enough to the basic orbiter to allow the basic + incremental method of calculation. Changes in vehicle mass properties will not affect aerodynamics, since aero forces and moments are referenced to a fixed aerodynamic reference point.

3.4.3 Rationale for Assumptions

Since no data was available for orbital flight aerodynamics for the NAR configuration at high angles of attack, it is assumed that the MDAC data are similar, since the vehicles are approximately the same shape.

REF. KEY

167 During orbital flight it is assumed that Newtonian (non-fluid) aerodynamics determine characteristics. This implies that only the surfaces of the vehicle facing the relative wind have any aerodynamic effects. The assumption that weather effects **diminish in importance at high altitudes is because of the low** dynamic pressure encountered in this regime. This also results in the negligible aerodynamic forces and moments just prior to orbit insertion.

The assumption that large roll excursions are not detrimental to the ascent trajectory is because pitch and yaw tradeoffs can be used to compensate for off-nominal roll.

3.4.4 Data References

167 Aerodynamic Design Data Book, Vol. I, Orbiter, Space Div., NAR, SD 72-SH-60-1, May, 1972.

168 Aerodynamic Design Data Book, Vol. II, Integrated Vehicle, Space Div., NAR, SD 72-SH-60-2, May, 1972.

59 Space Shuttle Orbiter Aerodynamics Data Book, McDonnell Douglas Delta Wing Design, MDAC, MSC, 1971.

REF. KEY

3.5 Ephemeris

3.5.1 Celestial Bodies

3.5.1.1 Requirements

Positions and velocities of several celestial bodies will be required for simulation of shuttle missions. Solar position relative to the vehicle will affect vehicle temperature distribution, star tracker resolution if pointed in that direction, as well as out-the-window views. Solar relative velocity will affect solar aberration. Since aberration (difference between apparent and real solar position due to spacecraft/sun relative velocity and the finite speed of light) could affect apparent position by 20 arc-seconds, it should probably be included. Positions of stars will be required for star-tracker and COAS use. The starfield will also be a significant part of the out-the-window view. Lunar position relative to the vehicle may affect the star-tracker as well as out-the-window views. Lunar phase may be of interest for improved realism, though there appears to be no situation (expect perhaps telescope sortie missions) in which it would affect crew or system response. Lunar aberration should be of the order of 5 arc-seconds, and, for the above - mentioned purposes, can probably be safely ignored. Positions of the bright planets (Venus, Mars, Jupiter) could conceivably affect the star tracker. In the case of sortie operations with attached telescope, it may also be desirable to know planetary positions. Pointing accuracies of less than 1 arc-minute are forseen under certain special circumstances. Thus, position errors for celestial bodies should be of considerably less magnitude than the above - particularly oscillatory errors deriving from such sources as slow computation rates. Star occultation by earth, sun, or moon has a significant effect on star tracker capabilities, and should be taken into account.

REF. KEY

3.5.1.2 Rationale for Assumptions

Not applicable.

3.5.1.3 References

1. 166 pp 3-160

3.5.2 Coordinate Transformations

3.5.2.1 Requirements

Transformations between Ephemeral coordinates and True-of-Date coordinates, and between True-of-Date coordinates and Geographic coordinates will be required. Each transformation changes in time. The transformation between Ephemeral and True-of-Date coordinates includes Earth precession and nutation effects. It need not be updated at a higher rate than once per 15 minutes; possibly no more than once per month. If True-of-Date is used as the prime reference system, vehicle state must be transformed at the time the system is updated to be consistent with the new system. The transformation between True-of-Date and Geographic coordinates is dependent only upon the true Greenwich hour angle at a given time. Since the hour angle changes at a rate of about .004 deg/sec, it must be updated frequently (1° hour angle error is equivalent at the equator to some 70 miles of ground track position error). Earth horizon position is required for use in the horizon scanners.

3.5.2.2 Rationale for Assumptions

Not applicable

3.5.2.3 References

Not applicable